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USAAVLABS TECHNICAL REPORT 64-68K

HEAVY-LIFT TIP TURBOJET ROTOR SYSTEM
VOLUME XI
ENGINE DESIGN

October 1965

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-25(T)

HILLER AIRCRAFT COMPANY, INC.
CONTINENTAL AVIATION AND ENGINEERING CORPORATION

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October 1965

HEAVY-LIFT TIP TURBOJET ROTOR SYSTEM

VOLUME XI

ENGINE DESIGN

CAE Report No. 942

Prepared for

Hiller Aircraft Company, Inc.

By

Continental Aviation and Engineering Corporation

For

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

(U. S. Army Transportation Research Command when report prepared)

PREFACE

This report, prepared by Continental Aviation and Engineering Corporation covers the design of the Continental Model 357-1 tip turbojet engine. This work was accomplished under Hiller purchase order HAC-1-64, Contract DA 44-177-AMC-25(T).

The Continental Model 357-1 tip turbojet is a modification of the J69-T-29 turbojet engine (CAE Model 356-7) and is designed to operate at the tip of a helicopter rotor.

This report is one of a series of three documents submitted to fulfill the requirements of Hiller purchase order HAC-1-64, Article 1 Statement of Work, Item (3) (e). The reports include:

"Continental Model 357-1 Tip Turbojet Engine - Engine Design," Heavy-Lift Tip Turbojet Rotor System, Volume XI, CAE Report No. 942, U.S. Army Transportation Research Command,* Fort Eustis, Virginia, October 1965.

"Continental Model 357-1 Tip Turbojet Engine - Fuel Pump and Control System Design," Heavy-Lift Tip Turbojet Rotor System, Volume XII, CAE Report No. 943, U.S. Army Transportation Research Command,* Fort Eustis, Virginia, October 1965.

"Continental Model 357-1 Tip Turbojet Engine - Preliminary Model Specification," Heavy-Lift Tip Turbojet Rotor System, Volume XIII, CAE Model Specification No. 2253, U.S. Army Transportation Research Command,* Fort Eustis, Virginia, October 1965.

*Changed to U. S. Army Aviation Materiel Laboratories in March 1965.

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SYMBOLS

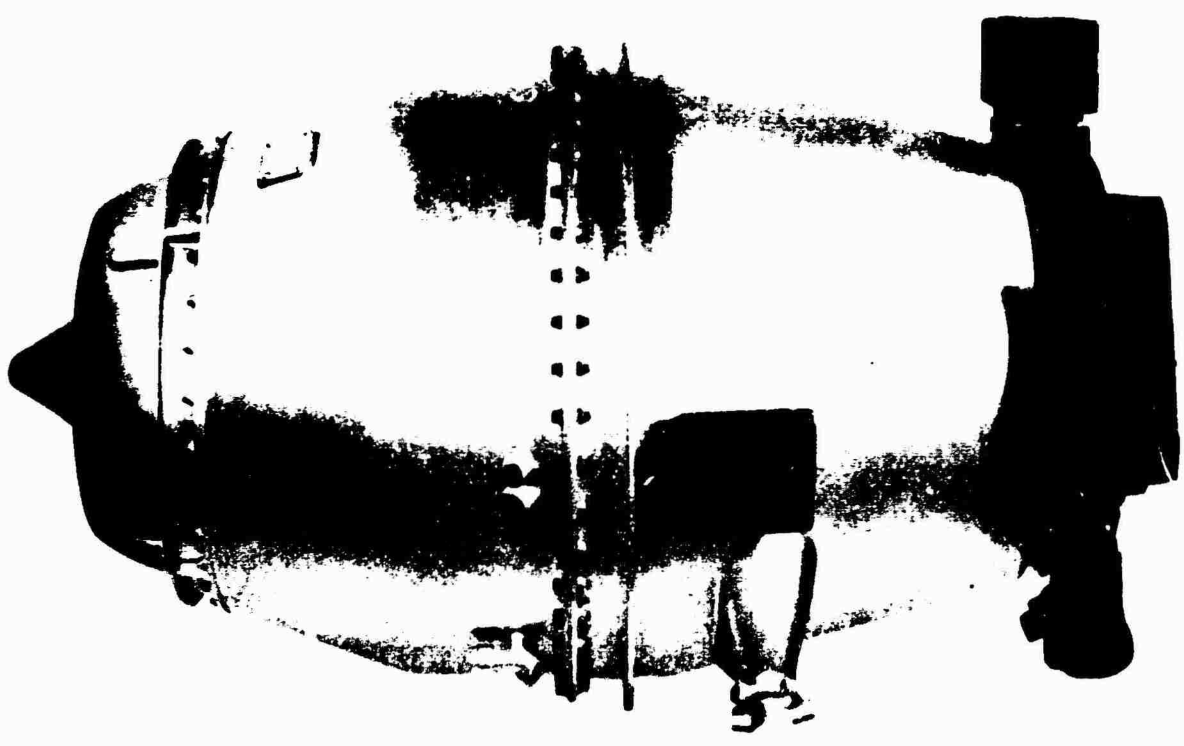
ω	engine speed
Ω	helicopter rotor speed - radians per second
$\dot{\phi}$	rotor blade pitching velocity - radians per second
g	acceleration field
σ	stress - pounds per square inch
F_n	net thrust - pounds
W_f	net thrust - pounds per hour
W_a	airflow - pounds per second
P_2/P_1	pressure ratio

SECTION ONE. SUMMARY

This report presents the design of the Continental Model 357-1 tip turbojet engine designed under Hiller Contract HAC-1-64.

The Model 357-1 is a modification of the J69-T-29 engine (Continental Model 356-7), designed to operate at the tip of a helicopter rotor.

Figure 1 shows the basic arrangement of the 357-1 engine. It incorporates a straight-through flow path with an integral jet nozzle with the minimum number of accessories necessary to operate the engine and features an integral oil tank and optional oil cooler, air starting, and provisions for an anti-iced intake.



RIGHT SIDE VIEW

Figure 1. Continental Model 357-1 Tip Turbojet Engine Mockup.

The static sea level performance targets for the fully qualified production Model 357-1 engine are summarized below:

Condition	Pounds Thrust	SFC
Military	1700	0.99
Normal	1375	0.98

The aerothermodynamic design of the engine is based on the proven J69 engine with an airflow of 29.5 pounds per second, 5.7 pressure ratio, and a turbine inlet temperature of 1560°F. The performance of the engine is shown in Figure 2.

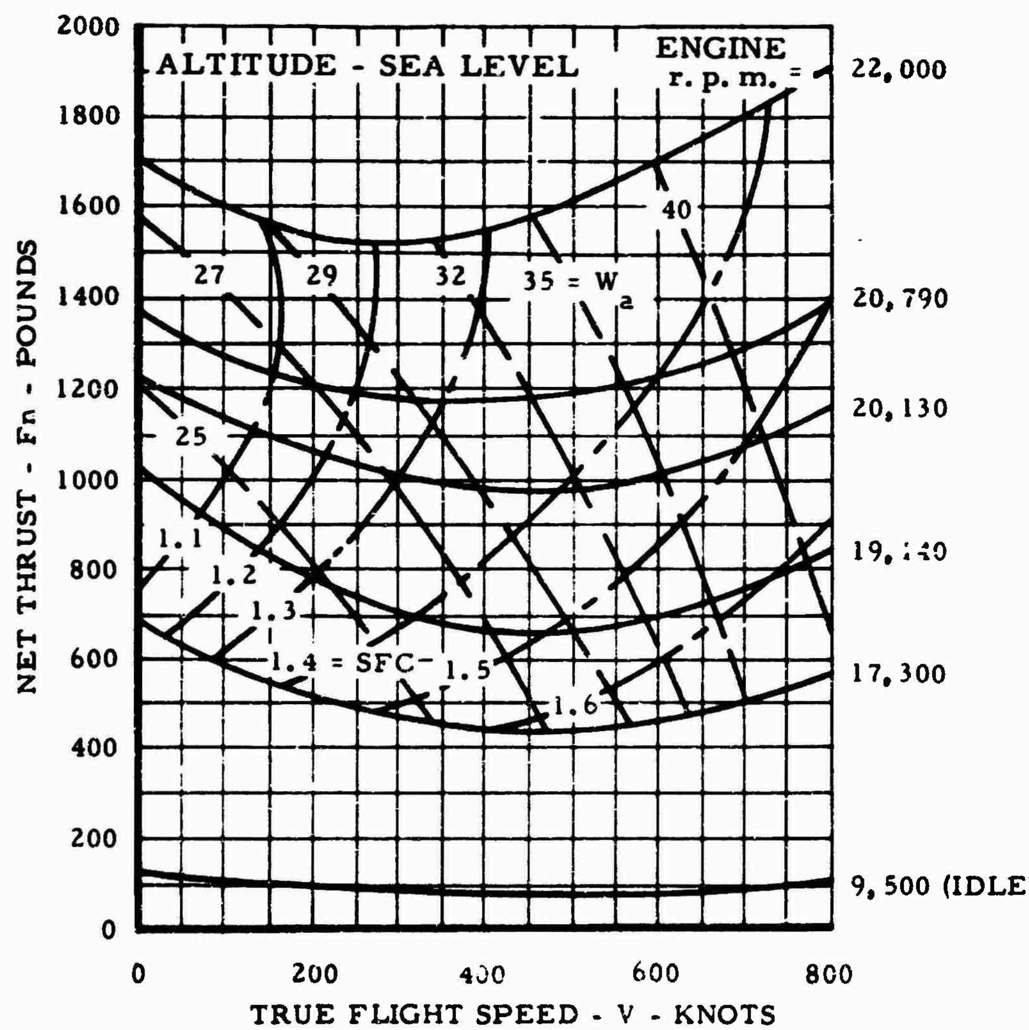


Figure 2. Continental Model 357-1 Tip Turbojet Performance Characteristics.

The complete engine, including oil tank and cooler, is 47.95 inches long with a maximum diameter of 27.80. The installation drawing is shown in Figure 3.

Preliminary design of the fuel control and fuel pump has been accomplished primarily to meet the requirements of, and to provide the necessary response and safety features for, a tip turbojet engine in the hover mode of operation.

Sufficient growth potential is available in the basic engine as both the low cycle temperature and the pressure ratio of the axial compressor can be increased. Increase in cycle temperature with the present compressor design would enable the engine to easily achieve a design point thrust of 1840 pounds at a specific fuel consumption of 1.1. A slightly modified axial compressor and a new combustor configuration will enable the engine to be rated at 1900 pounds thrust at 1.12 specific fuel consumption. Both of these aerodynamic configurations have already been tested and the performance verified. Further increases in performance could be realized by aerodynamic development of the centrifugal compressor and a new axial compressor designed for a higher pressure ratio and mass flow. Without adding stages, it should be possible to develop this basic engine to 2000 pounds of thrust at approximately 1.1 specific fuel consumption. These modifications plus the possibility of adding stages, which would give as high as a twenty-five percent increase in thrust, should enable the basic engine to meet higher horsepower requirements.

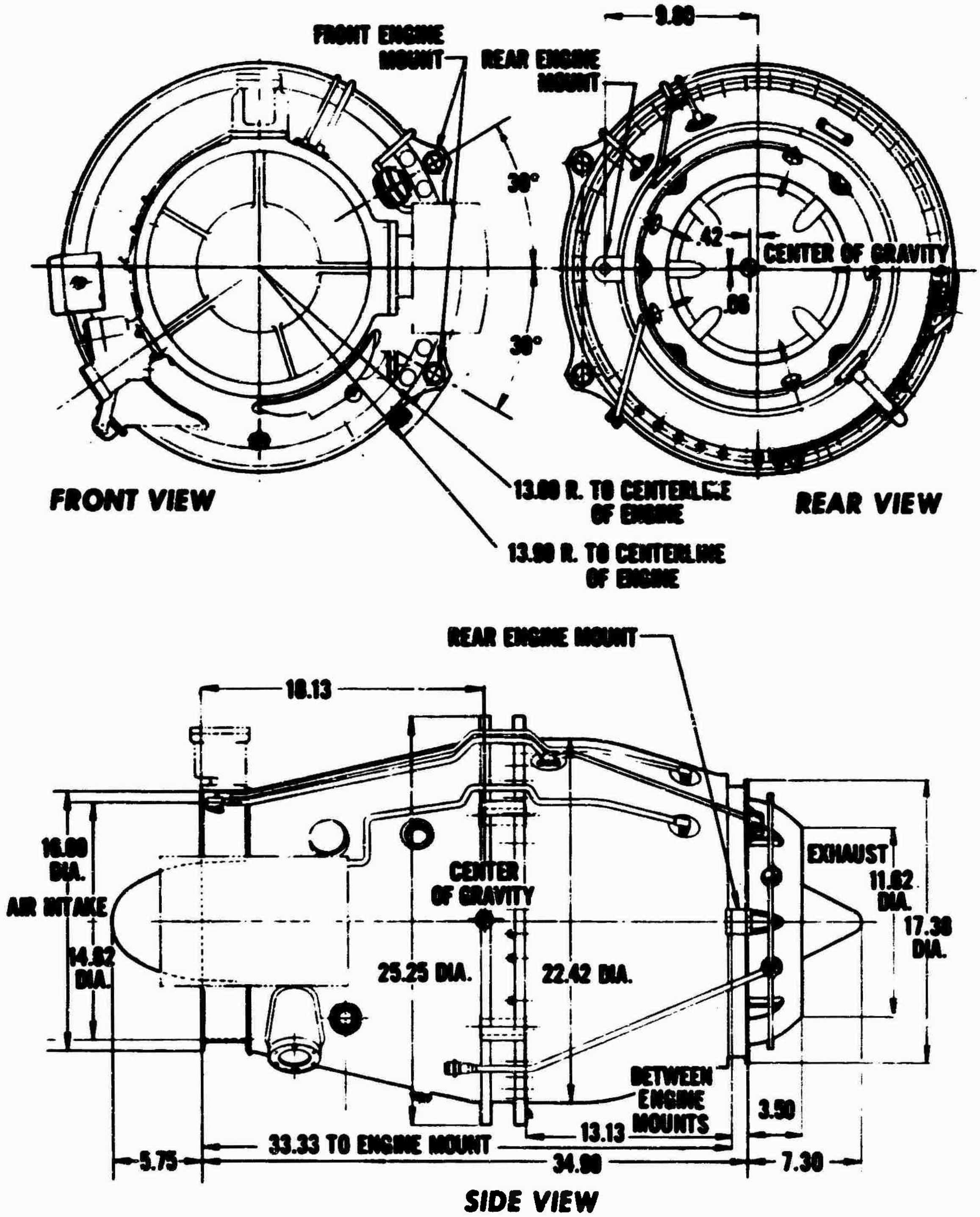


Figure 3. Continental Model 357-1 Tip Turbojet Engine Installation Drawing.

SECTION TWO. INTRODUCTION

The CAE Model 357-1, Figure 4, is a turbojet engine rated at 1700 pounds static sea level thrust, designed specifically for rotor tip application.

The engine employs the straight-through aerodynamic flow path of the Continental J69-T-29 engine. Air enters an annular intake at the front of the engine and passes successively through a single-stage transonic axial compressor, a two-stage stator, a single-stage centrifugal compressor, radial and axial diffusers, an annular combustor, a turbine inlet nozzle, a single-stage turbine, and an integral fixed-area jet nozzle.

The engine utilizes a maximum diameter structural shell with the two primary mounting points near the center of gravity and a steady rest mount at the rear of the engine.

The engine employs a circulating lube system with an integral oil tank and optional engine-mounted oil cooler. The lube system is designed to operate in the 1g vertical field static and a 235g horizontal field peculiar to tip turbojet operation.

OPERATING ENVIRONMENT

The operating environment for a turbojet engine mounted at the tip of a helicopter rotor is unique. The engine is subjected to continuous loading of considerable magnitude, both static and alternating, not encountered in any other application. The three basic load directions are shown in Figure 5. The direction "R" is normal to the helicopter rotor axis and directed radially outward, and is the direction of the predominant load. The direction "F" is tangent to a circle drawn through the tip of the helicopter rotor blades. The direction "V" is parallel to the helicopter rotor axis. There is a continuous g field in the "R" direction with a superimposed alternating g field. The g fields in the "F" and "V" directions are alternating. A summary of the g fields in the three basic directions is presented in Table 1.

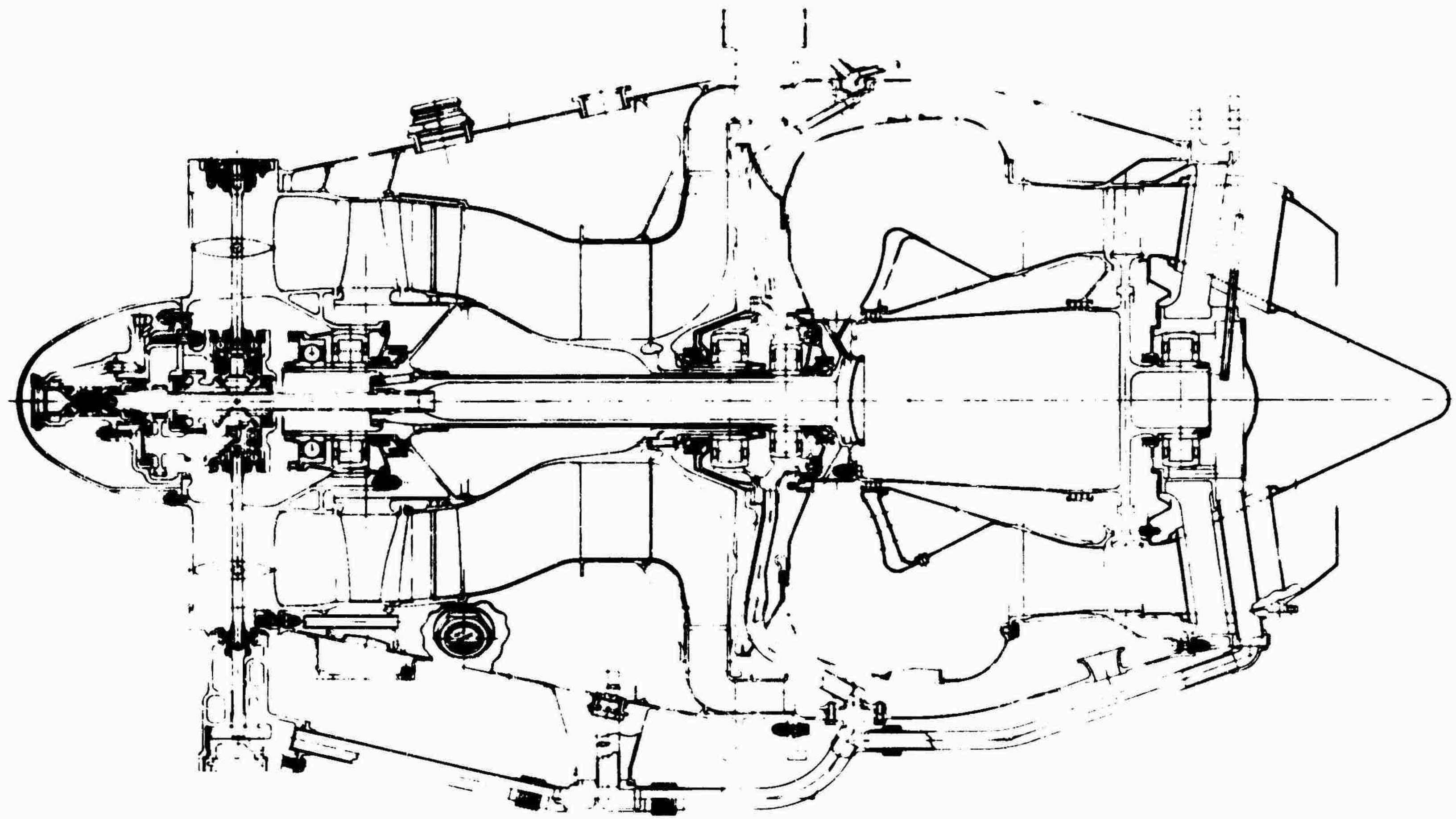


Figure 4. Continental Model 357-1 Tip Turbojet Engine Cross Section.

TABLE 1
TIP TURBOJET g FIELD SUMMARY

HOVERING

Direction	Continuous	Alternating	Frequency*	Percent of Time
R	235	± 10	1	5
		± 20	0.2	2
		± 1	6	1
F	-	± 5	1	5
		± 2	0.2	2
		± 3	6	1
V	1	± 40	1	5
		± 20	1	2
		± 10	0.67	1

CRUISE

Direction	Continuous	Alternating	Frequency*	Percent of Time
R	195	± 10	1	5
		± 20	0.2	2
		± 1	6	1
		± 5	1	50
F	-	± 5	1	5
		± 2	0.2	2
		± 3	6	1
		± 3	3	50
V	1	± 40	1	5
		± 20	1	2
		± 10	0.67	1
		± 15	2	49
		± 25	2	1

* Per revolution of helicopter rotor.

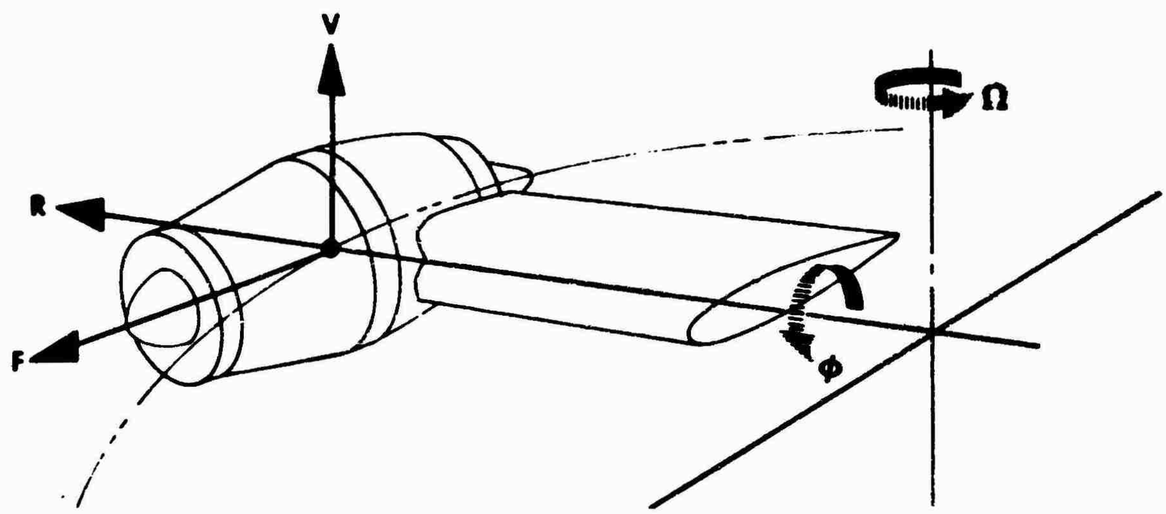


Figure 5. Engine Load Directions.

The hovering g field is applicable for 7 percent of the operating time and the cruise g field for 93 percent of the operating time. The engine is capable of an overspeed to 259g for a total time of 30 minutes per 1,000 hours of total flight time with one minute per occurrence.

The angular velocity summary is presented as follows:

In plane velocity Ω

Hovering flight (7 percent):	Normal operation 10.8 radians per second. Overspeed 11.3 radians per second for 1 minute per occurrence for a total of 30 minutes per 1,000 hours of total flight time.
Cruise flight (93 percent):	Normal operation 9.8 radians per second. Harmonic variation of $\pm$ 0.25 radians per second at a frequency of 1 per helicopter rotor revolution for 50 percent of the cruise flight time.

Pitching velocity $\dot{\Phi}$

50 percent of cruise flight:	$\pm$ 1 radian per second at a frequency of 1 per helicopter rotor revolution.
Maneuvering flight:	$\pm$ 3 radians per second at a frequency of 1 per helicopter rotor revolution for 2 percent of the total flight time.

The complete operating environment specified in MIL-E-5007B are applicable except that the g fields and operational attitudes peculiar to operation at the tip of a helicopter rotor shall apply.

The operation at the tip of a helicopter rotor results in high rotor bearing loads due to the centrifugal field, a continuous gyroscopic couple, and an alternating Coriolis force parallel to the engine rotor axis. A simplified diagram illustrating these forces are shown in Figure 6. Due to rotation ω about the engine axis, any point "P"

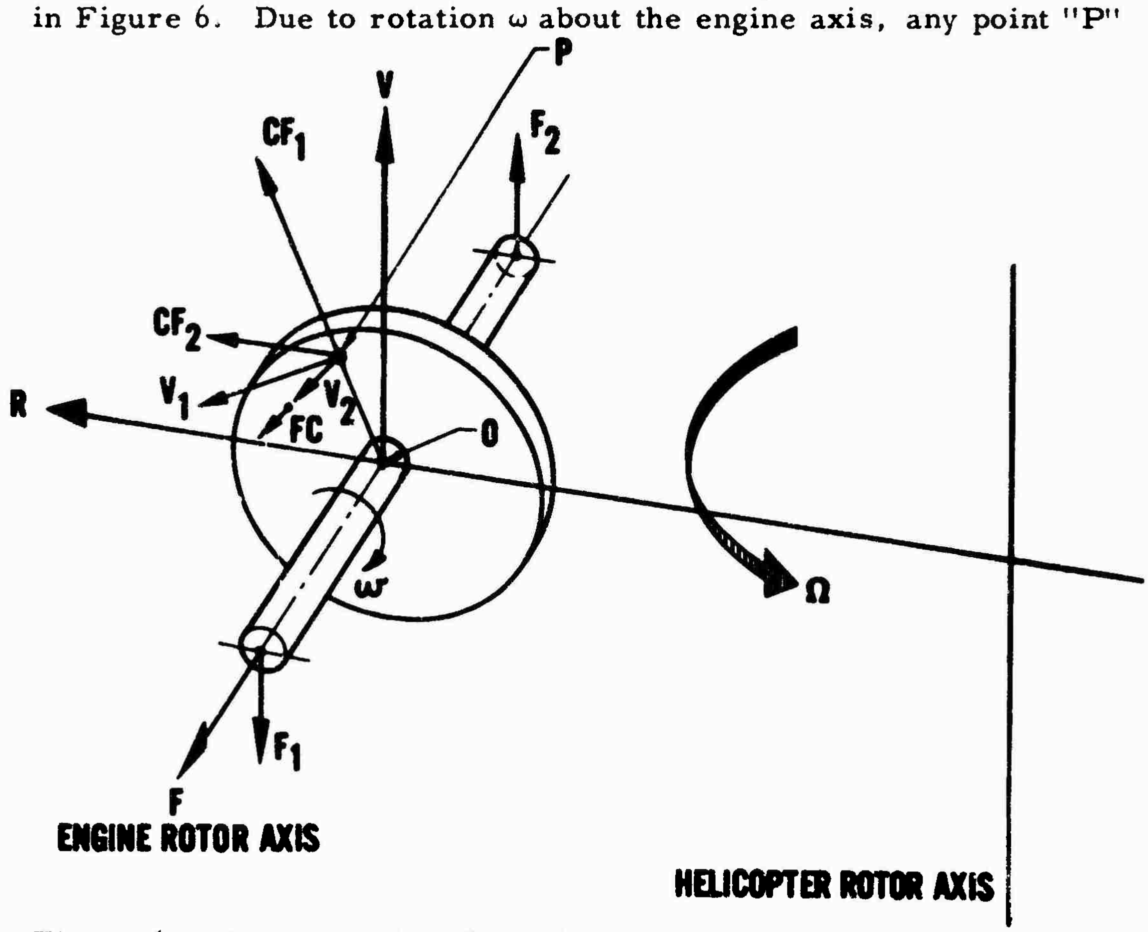


Figure 6. Forces Acting Upon Gas Turbine Rotating Elements.

knows a constant velocity V_1 and a constant centrifugal force CF_1. Due to rotation Ω about the helicopter axis, the same point "P" knows a variable velocity V_2 and a variable centrifugal force CF_2. Due to the variation in velocity V_2 in the direction parallel to the engine axis, point "P" knows an acceleration force FC, commonly known as the Coriolis force. The summation of the Coriolis forces produce a gyroscopic couple with resulting reactions F_1 and F_2. The Coriolis force, FC, changes direction above and below the "OR" axis. The alternating Coriolis force, FC, produces alternating stresses in all rotating blades, discs, and shafts with a frequency of one cycle per engine revolution.

SECTION THREE. MECHANICAL DESIGN

AERODYNAMIC FLOW PATH

The Continental Model 357-1 engine incorporates a straight-through aerodynamic flow as shown in Figure 7. The flow path incorporates a minimum number of parts and the least amount of aerodynamic obstructions to ensure minimum cost and maximum performance. Air enters the engine axially through a cast aluminum housing (No. 1, Figure 7). From the inlet housing the air passes through a 1.6:1 pressure ratio transonic axial compressor rotor (2), which is machined from a titanium forging. The air from the axial compressor is diffused through a two-stage, precision-cast aluminum stator (3). From the stators the air is directed into a centrifugal compressor (4), which is composed of a machined titanium inducer and a machined titanium radial impeller. From the centrifugal compressor stage the air passes through a fabricated stainless steel radial (5) and axial (6) diffuser. From the axial diffuser the air flows through the combustor housing (7) to the annular combustor (8) where fuel is added and combustion takes place. From the combustor the hot gases enter the turbine inlet nozzle (9) which is fabricated from precision-cast vanes and forged, fully machined shrouds. The gases then pass through a forged, fully machined turbine (10). From the turbine the gases then exit through the short integral jet nozzle (11).

ROTOR SYSTEM

The Model 357-1 rotor group, Figure 8, has two independently supported shaft assemblies, coupled by a relatively flexible extension of the turbine shaft.

The compressor shaft assembly consists of an axial compressor (1) and inducer (2) weldment, on which a radial impeller (3) is piloted and attached. The axial compressor-inducer shaft is made of two parts and electron beam welded. The intermediate section between the compressor and inducer has been aerodynamically contoured to form the inner boundary of the airflow path. The axial compressor is cantilevered from the shaft and extends partially over the front bearing journal (4).

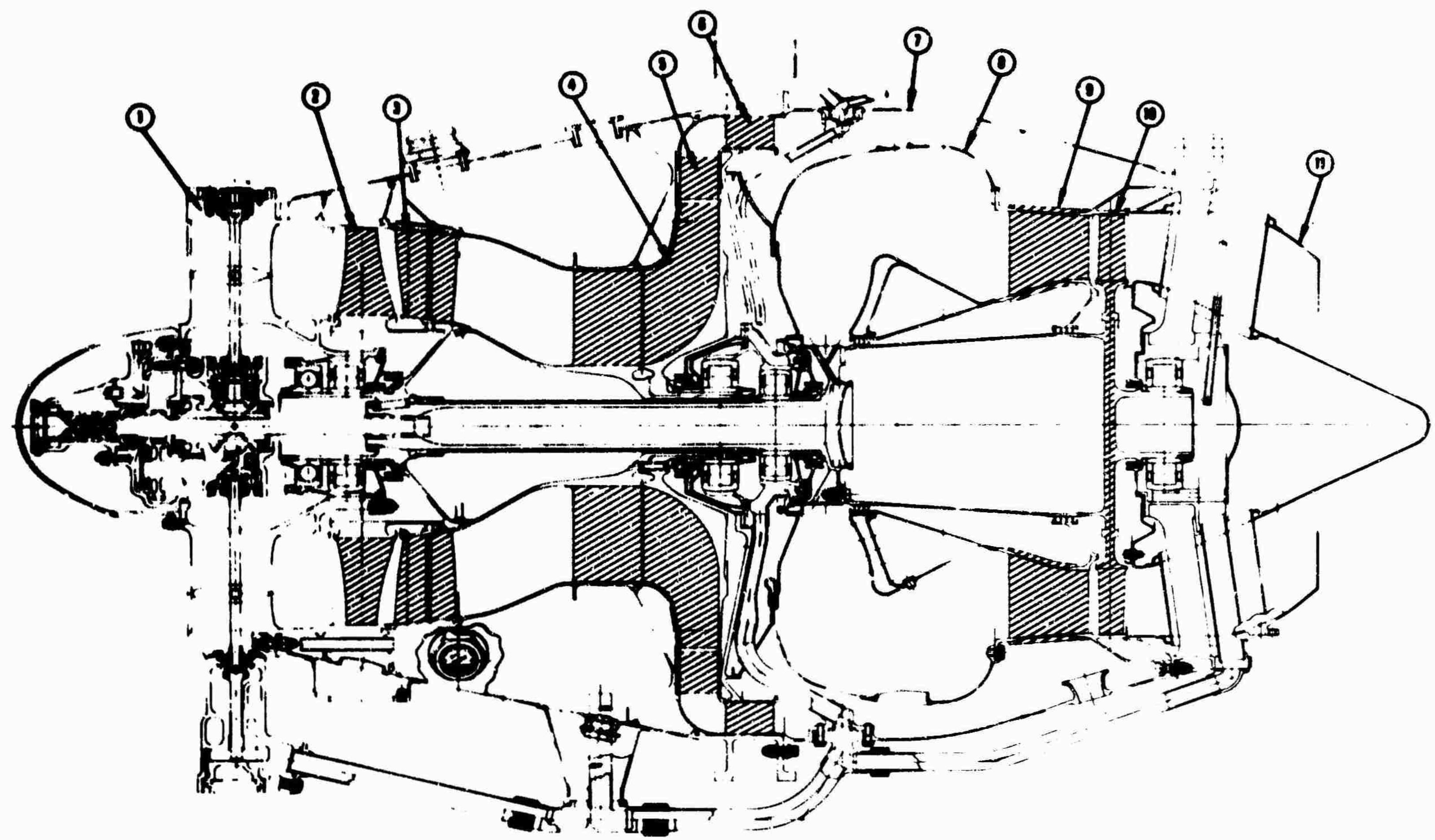


Figure 7. Aerodynamic Flow Path - Continental Model 357-1

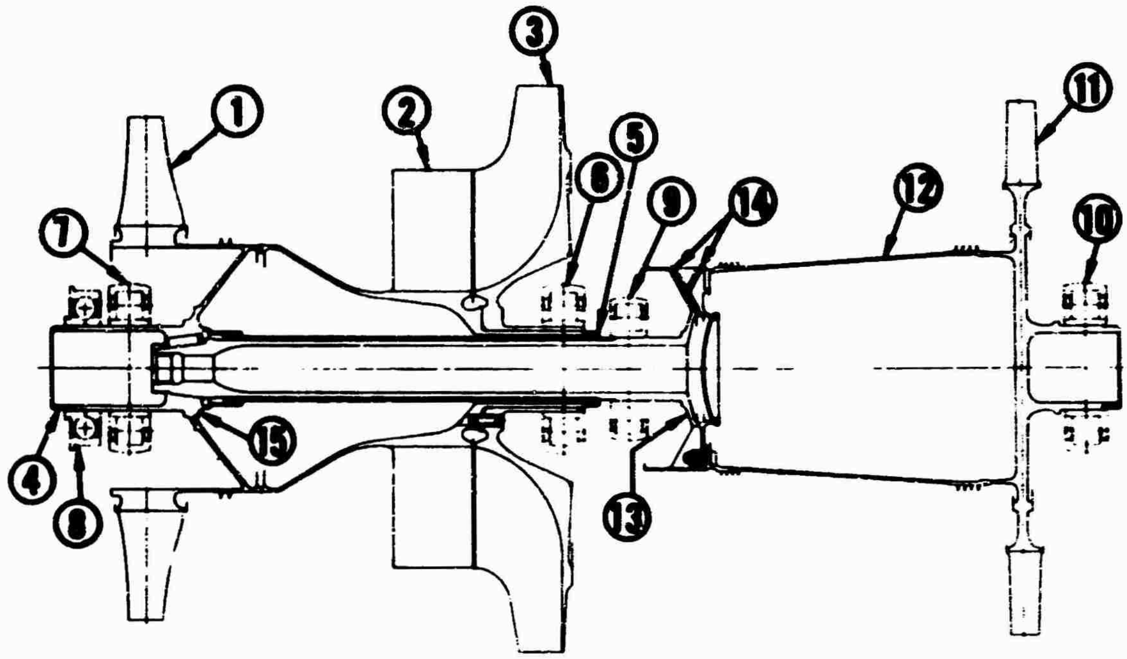


Figure 8. Rotating Assembly - Continental Model 357-1.

The turbine shaft assembly consists of a turbine rotor (11) and intermediate shaft (12) electron beam weldment to which a shaft extension (13) is bolted and piloted. The shaft extension is coupled to the inner hub (15) of the compressor shaft. The extension is designed to transmit torque and axial load from the turbine to the compressor shaft and to be relatively flexible in bending. The turbine shaft extension also serves as a fuel transfer tube, the fuel slinger (14), and the turbine shaft forward bearing journal. The rear turbine shaft journal is an extension of the turbine rotor.

ROTOR SUSPENSION

The compressor shaft assembly is straddle-mounted between two roller bearings. The rear bearing (6) is mounted on the radial compressor extension (5) and is housed in the forward section of the center support housing. The front bearing (7) is axially located so the compressor shaft radial load is divided approximately equal between the front and rear bearings.

The engine thrust bearing (8) is mounted on the compressor shaft journal, forward of the front radial bearing. Both the thrust bearing and the compressor shaft front bearing are mounted in a steel housing, which is piloted and bolted to a cantilevered section of the front bearing support. The thrust bearing mounting may be made with a controlled radial spring rate by cutting a number of longitudinal slots in the steel housing. The soft suspension of the thrust bearing will ensure proper load sharing between the two front bearings; that is, the thrust bearing will absorb all the axial load and only a small portion of the radial load.

The turbine shaft assembly is supported at the center and rear by roller bearings and at the front by the compressor shaft. The forward turbine shaft bearing (9) is housed in the rear section of the center support housing. The rear bearing (10) is housed in the engine rear bearing support.

The roller bearings are mounted in spherical housings that permit the bearings to align themselves in the direction of the deflected or misaligned shaft. This mounting arrangement prevents edge loading the rollers and minimizes prestressing the shafts.

The main shaft bearings have been specifically designed for the helicopter rotor tip application. The roller bearings are manufactured from consumable electrode vacuum melt (CEVM) M-50 tool steel with a minimum hardness of Rockwell C-60. The thrust bearing is manufactured from CEVM 52100 steel with a minimum hardness of Rockwell C-58. The use of vacuum melt results in higher quality steels. M-50 tool steels have hardness qualities superior to normal 52100 steels, particularly in high temperature applications. These improved quality and/or hardness properties have resulted in bearing steels with fatigue life several times that calculated for standard bearings. This increase in life expectancy is indicated in the bearing load-life tabulation.

The rotor thrust bearing is a 60 mm bore, split inner race, radial contact groove ball bearing. The bearing has a complement of 14 balls, 5/8-inch in diameter, having a normal operating contact angle of 25 degrees. The inner and outer race curvatures are 52 and 53 percent, respectively. The bearing has a special flanged outer race that is piloted and clamped in the housing. The single-piece bearing cage is machined from AMS 6415 steel, heat treated to Rockwell C-34 maximum, and silver plated 0.001- to 0.002-inch thick. The cage is outer-race land piloted. The pockets of the cage are flatted in the axial direction to minimize Coriolis loading by the balls. The ball component is trapped in the cage by deformed tangs on the outer diameter. The cage-ball assembly is removable from the outer race. The front half of the inner race has a puller groove. The gearing is manufactured to ABEC-5 standards.

Three of the rotor radial bearing are 60 mm bore roller bearings and one is a 45 mm bore roller bearing. These bearings have wide inner rings without lands. The rollers are accurately guided by lands on the outer race. The bearings have a one-piece, machined, silver plated, steel cage which is piloted by the inside diameter of the outer race lands. The rollers, cage, and outer race are nonseparable. The outside diameter of the outer race is ground spherical to close dimensional and concentricity control. The 60 mm bore roller bearings have twelve 20 mm rollers and the 45 mm bore roller bearing has ten 19 mm rollers. The bearings are manufactured by RBEC-5 standards.

ROTOR BEARING ANALYSIS

The tip turbojet environment produces a severe, but not wholly unknown, combination of bearing loads. The three types of loads are:

1. Radial Loads
2. Continuous g Field Operation
3. Continuous Gyroscopic Forces

Current aircraft engines with planetary or epicyclic reduction gearing have bearings operating under radial loads in a continuous g field. The g field in some cases is greater than the tip turbojet environment.

Current aircraft engine bearings are subjected to intermittent gyroscopic forces resulting from aircraft flight and ground maneuver turns. The gyroscopic forces result in overturning couples on the bearing retainer and rolling elements. The effects of these gyroscopic couples, although not amenable to analytical evaluation, should not prove insurmountable to develop.

The main bearing radial reactions are the result of the acceleration forces on the engine rotor in the radial g field and the gyroscopic field. These reactions, centrifugal and Coriolis, act in planes

90 degrees to each other. The steady-state radial loads have been calculated on a duty cycle basis of 93 percent operation at normal rating (20, 790 engine r.p.m. and 640 feet per second helicopter rotor tip velocity) and 7 percent operation at take-off or hover (22, 000 engine r.p.m. and 700 feet per second helicopter rotor tip velocity). The equivalent load is based upon the above duty cycle in proportion to the roller bearing life function.

The basic dynamic load ratings of the roller bearings have been calculated using AFBMA standards. The basic load rating is 30, 200 pounds per million revolutions for the 60 mm roller bearing and 23, 200 pounds per million revolutions for the 45 mm roller bearing.

The thrust bearing load is assumed to be constant (1270 pounds) at both normal and take-off rated conditions. The bearing load is assumed to be pure thrust and, therefore, the equivalent load is 1270 pounds. The basic dynamic load rating is 10, 580 pounds per million revolutions, using AFBMA standards.

The equivalent rotor speed of 20, 860 r.p.m. has been calculated as the sum of the products of time and speed at take-off and time and speed at rated engine power.

Table 2 presents the load and life summary for the main shaft bearings.

TABLE 2
MODEL 357-1 MAIN SHAFT BEARING SUMMARY

Function	Thrust Bearing	Front Comp. Roller Bearing	Rear Comp. Roller Bearing	Forward Turbine Roller Bearing	Rear Turbine Roller Bearing
Cent. Load at 235g		3842	3902	2500	3321
Cent. Load at 195g		3188	3238	2075	2756
Equiv. Cent. Load		3233	3284	2105	2796
Gyro. Load at 235g		2278	-2533	2083	-1827
Gyro. Load at 195g		1971	-2191	1801	-1580
Equiv. Gyro. Load		1992	-2215	1821	-1597
Equiv. Load (pounds)	1270	3797	3961	2783	3220
Materials Life Factor	5	10	10	10	10
B_{10} Life Rating (hours)	2315	7980	6950	9350	13,720

ROTOR STRUCTURAL ANALYSIS

The Model 357-1 engine rotor is subject to all the forces normally encountered in a jet engine plus the acceleration forces produced by the engine masses rotating about the helicopter rotor centerline.

The engine rotor is subject to two significant primary stresses normally encountered in any jet engine. These stresses are:

1. The centrifugal stresses due to rotation about the engine axis.

2. An alternating stress due to circumferential variations in aerodynamic loadings.

The frequency of these alternating stresses is generally a function of strut and stator vane spacing adjacent to the rotating part. Rotation about the helicopter rotor centerline produces two significant accelerations, which are termed secondary in order to differentiate them from the normal engine acceleration forces. These accelerations are:

1. The centrifugal field.

2. The Coriolis or gyroscopic forces and moments.

The centrifugal field is relatively constant and is directed radially outward from the helicopter rotor centerline. The engine rotor blading changes orientation in the centrifugal field once per engine revolution and experiences an alternating stress.

The Coriolis forces are variable in magnitude and are directed parallel to the engine axis. The rotor blading, discs, and shafts experience an alternating stress due to these alternating forces.

The two secondary stresses are alternating in nature with a frequency of once per engine revolution.

A typical blade stress graph is shown in Figure 9, illustrating the mean blade stress, the cyclic gas bending, and the cyclic Coriolis. The mean stress shown is a summation of the P/A stress due to the primary centrifugal field and any steady-state gas bending stresses. The effects of blade twist and the restoring force due to the primary

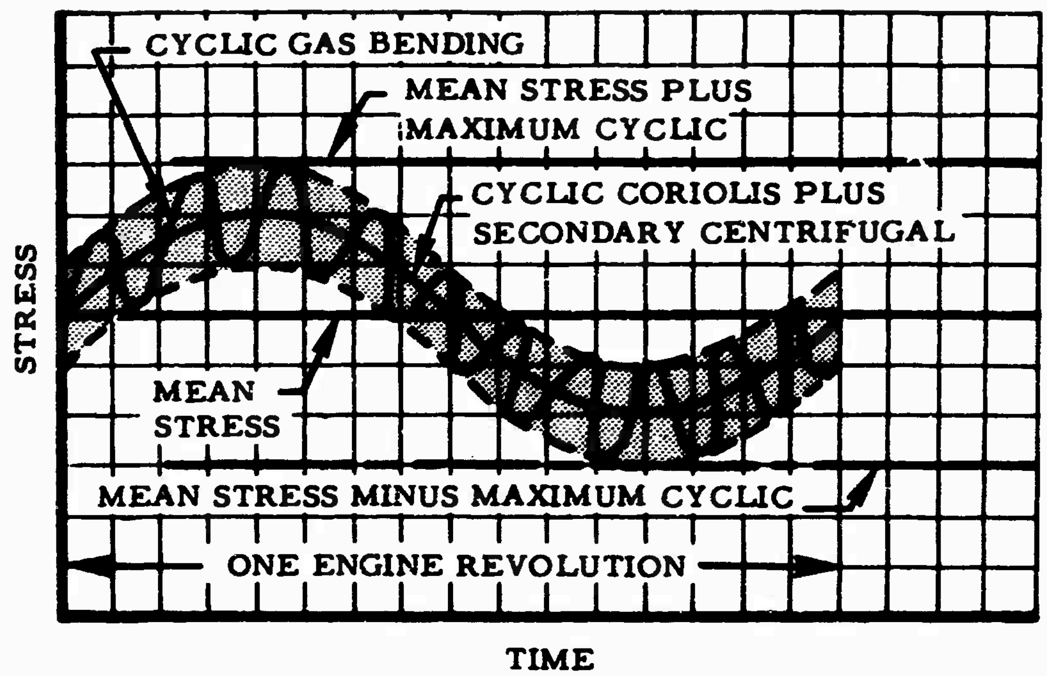


Figure 9. Typical Blade Stress - Continental Model 357-1 Engine.

centrifugal field has been fully considered in the blade stress analysis.

Particular emphasis has been placed upon providing generous airfoil root fillets. These fillets have been designed to conform to an ideal elliptical shape to allow a smooth load transition from the blade to the rim with low stress concentration.

Axial Compressor

The axial compressor is an integrally bladed rotor, fully machined from a titanium forging. The blade mean stress and cyclic stress summary is shown in Figure 10, and the interference diagram in Figure 11. The modified Goodman diagram is shown in Figure 12; the allowable cyclic stress is 85 percent of the 10^8 endurance limit, and the allowable mean stress is 85 percent of the 0.2 percent yield.

The tangential stresses in the rim section vary slightly in the axial direction; the maximum stress is 52,600 p. s. i.

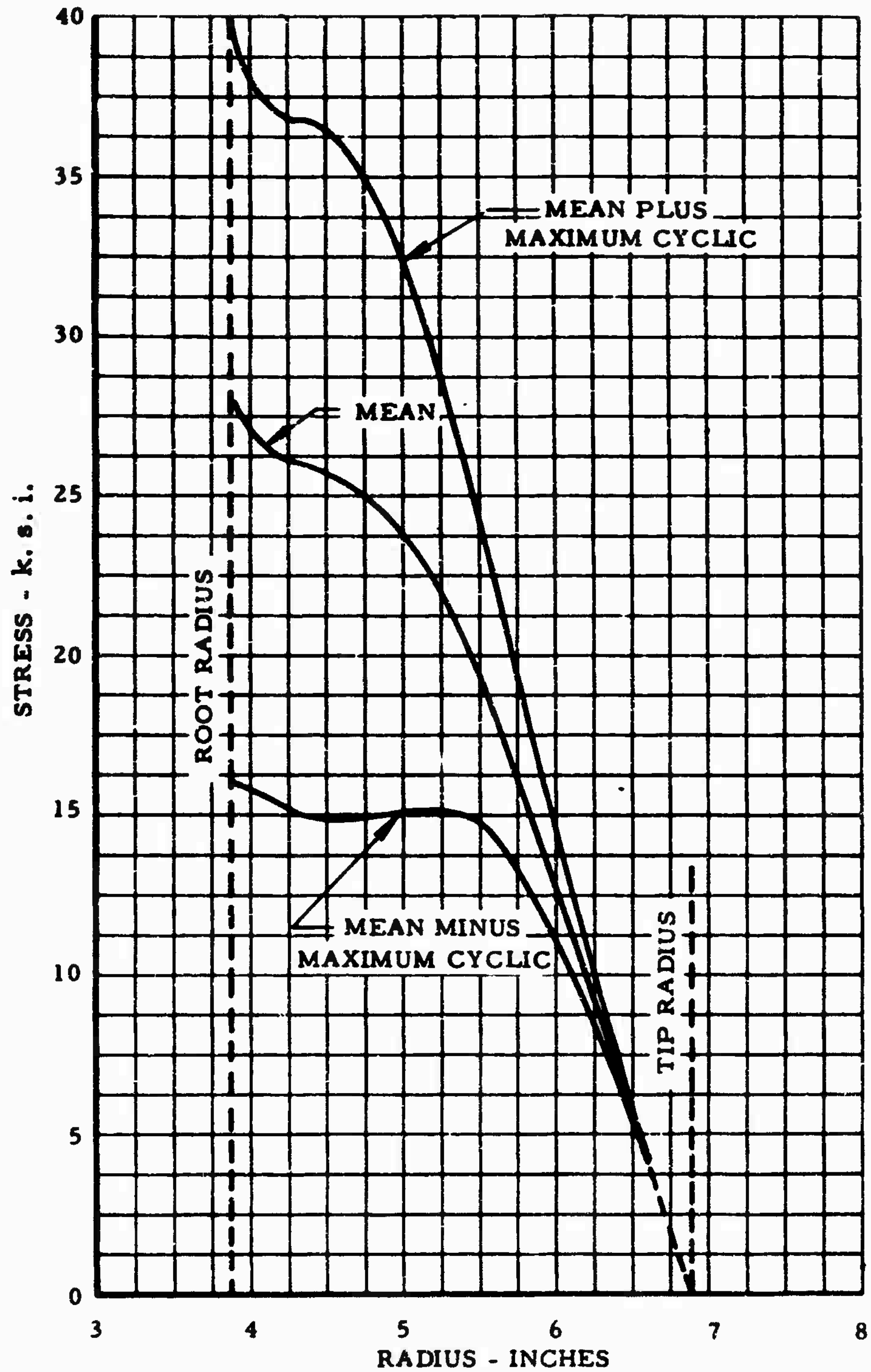


Figure 10. Axial Compressor Blade Stress Summary - Continental Model 357-1 Engine.

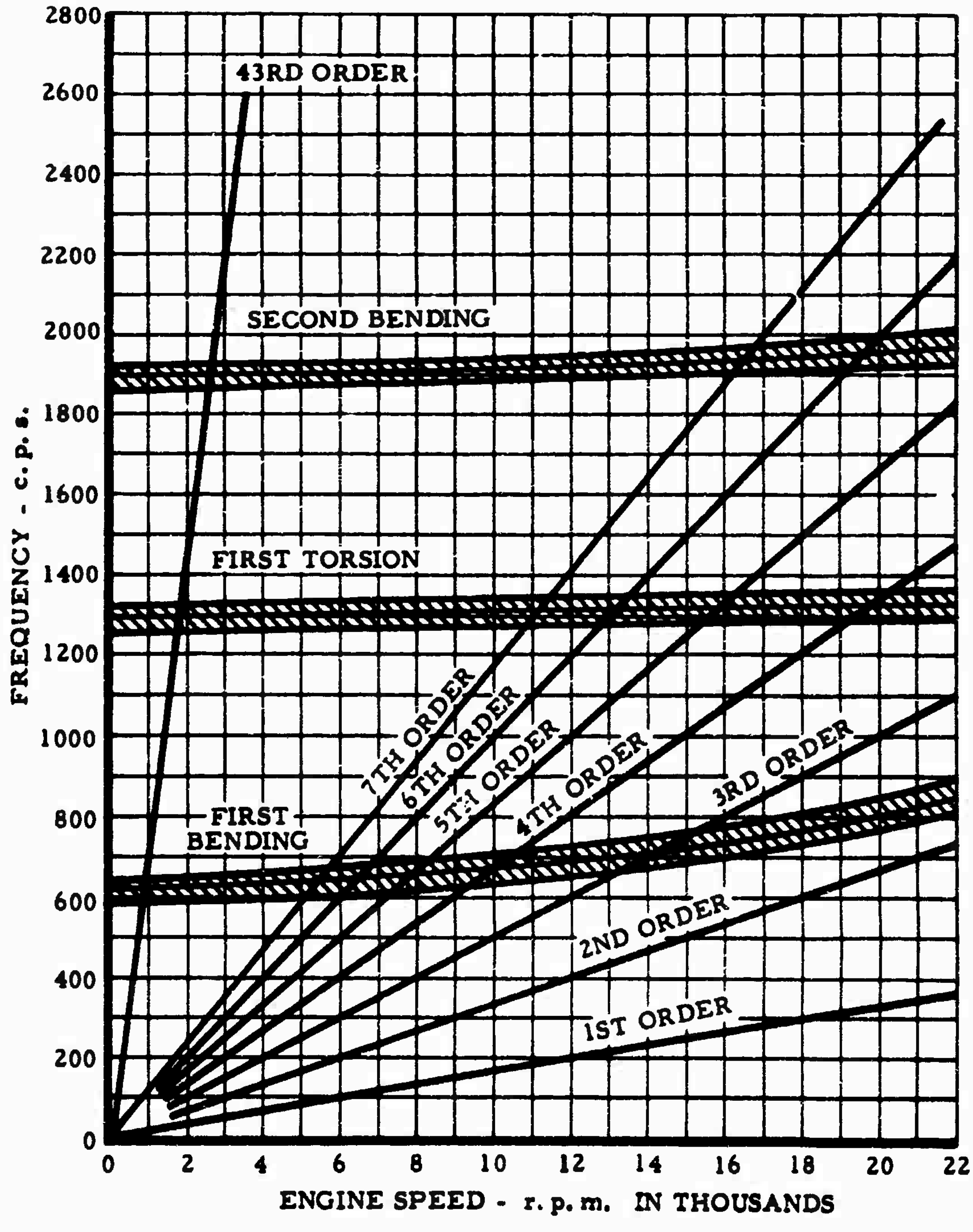


Figure 11. Axial Compressor Blade Interference Diagram - Continental Model 357-1 Engine.

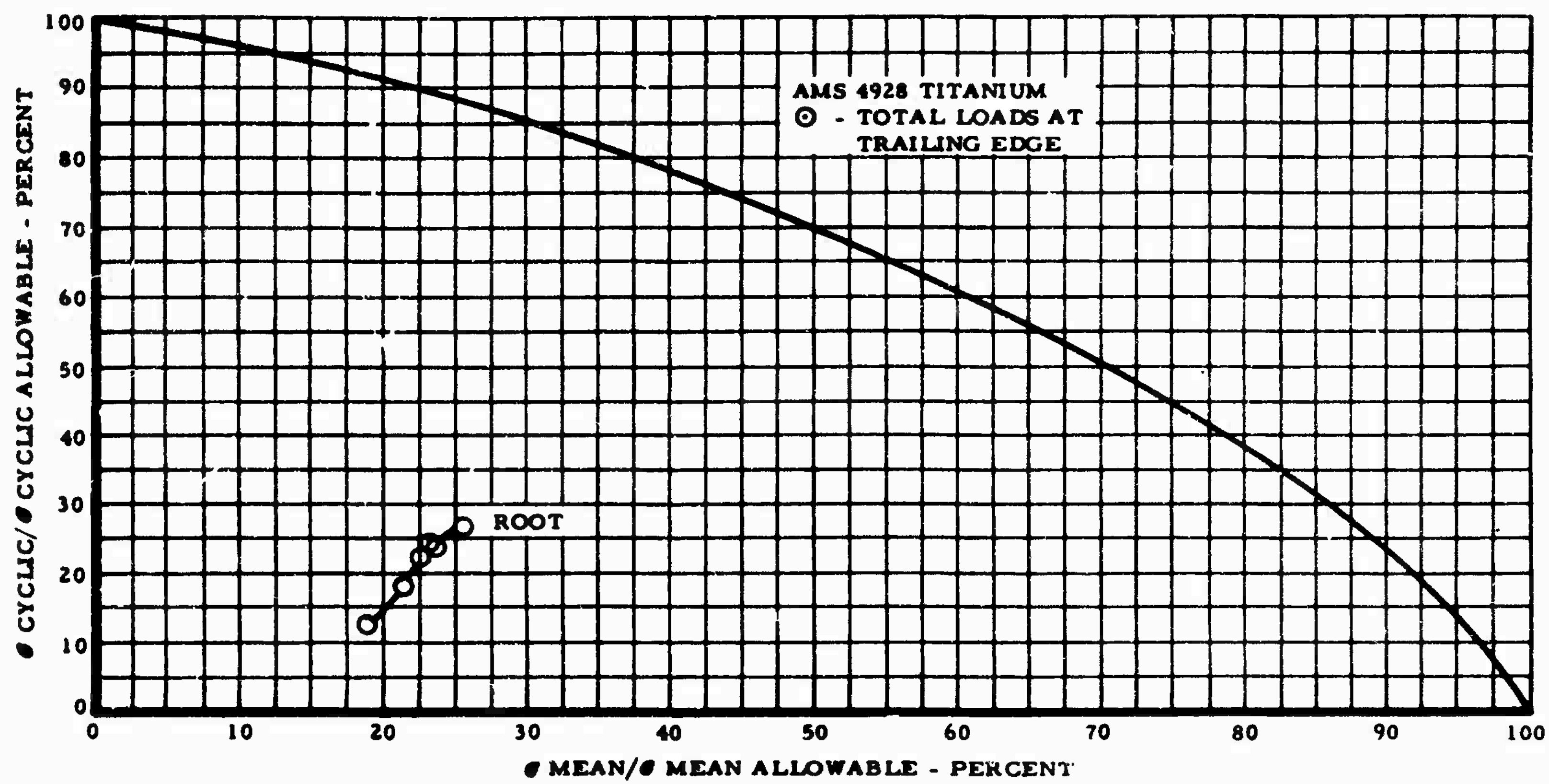


Figure 12. Modified Goodman Diagram for Continental Model 357-1 Axial Compressor Blade.

The material is AMS 4928 titanium with an ultimate of 130,000 p.s.i., a yield of 120,000 p.s.i., and an endurance limit of 53,000 p.s.i. with adequate structural margin.

Inducer Compressor

The inducer compressor is an integrally bladed rotor, fully machined from a titanium forging. The blade mean stress and cyclic stress summary is shown in Figure 13, and the blade interference diagram is shown in Figure 14. The modified Goodman diagram is shown in Figure 15; the allowable cyclic stress is 85 percent of the 10^8 endurance limit, and the allowable mean stress is 85 percent of the 0.2 percent yield.

The maximum stresses in the rim are 46,500 p.s.i., tangential; and 28,000 p.s.i., radial.

The material is AMS 4928 titanium with an ultimate of 130,000 p.s.i., a yield of 120,000 p.s.i., and an endurance limit of 53,000 p.s.i with adequate structural margin.

RADIAL COMPRESSOR

The radial compressor is an integrally bladed rotor fully machined from a titanium forging. The blade stress summary is shown in Figure 16. The blade interference diagram in Figure 17 is based on experimental data. The disc stress summary is shown in Figures 18 and 19. The modified Goodman diagram for the blade and disc is shown in Figure 20.

The material is 6AL-6V-2Sn with an ultimate of 150,000 p.s.i., a yield of 140,000 p.s.i., and an endurance limit of 62,500 p.s.i. with adequate structural margin.

Turbine

The turbine is an integrally bladed rotor fully machined from a nickel alloy forging. The blade mean stress and cyclic stress summary is shown in Figure 21, and the blade interference diagram in Figure 22. The modified Goodman diagram for the blade is shown in Figure 23; the allowable cyclic stress is 85 percent of the endurance limit and the allowable mean stress is 85 percent of the 1,000-hour

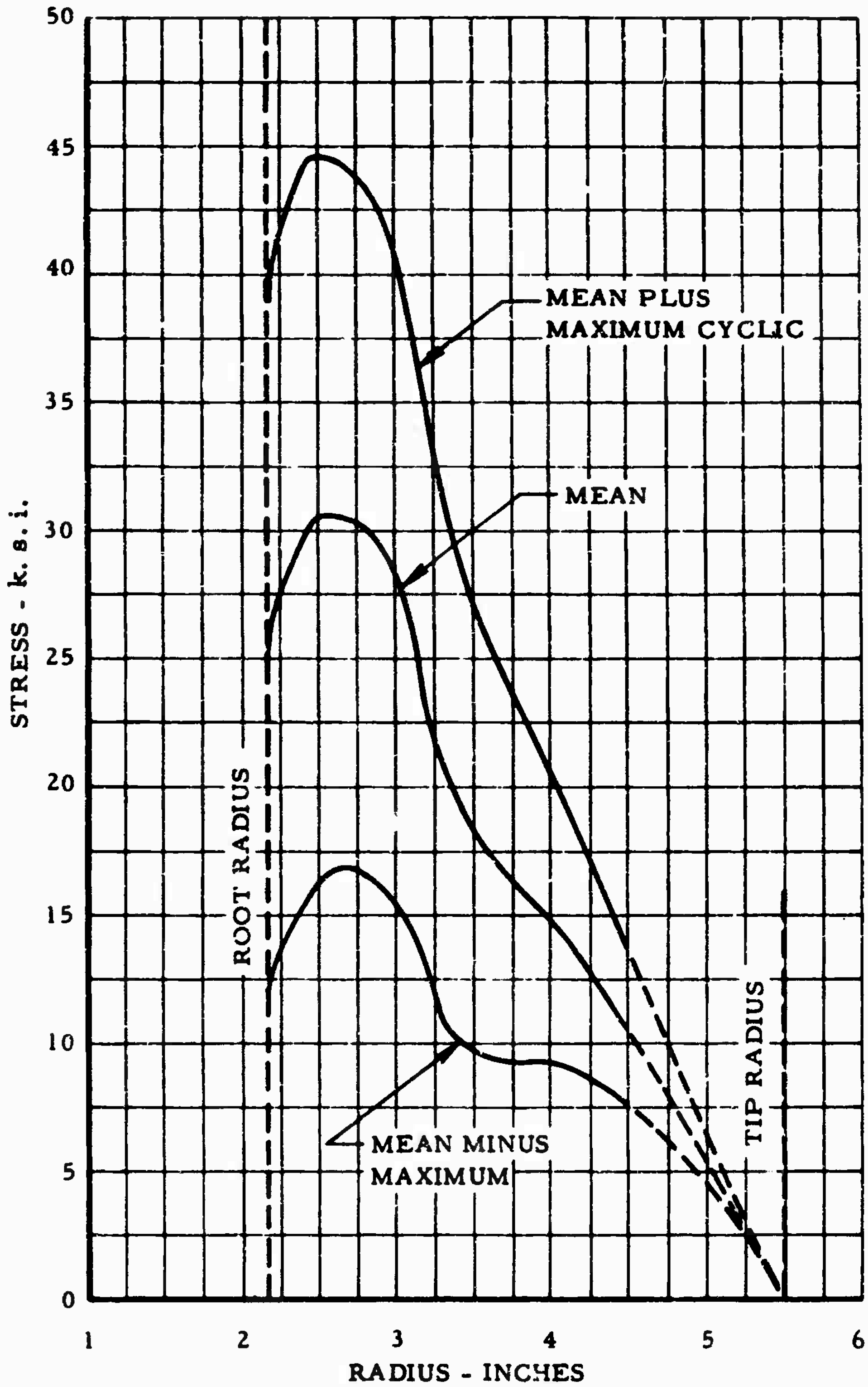


Figure 13. Inducer Blade Stress Summary - Continental Model 357-1 Engine.

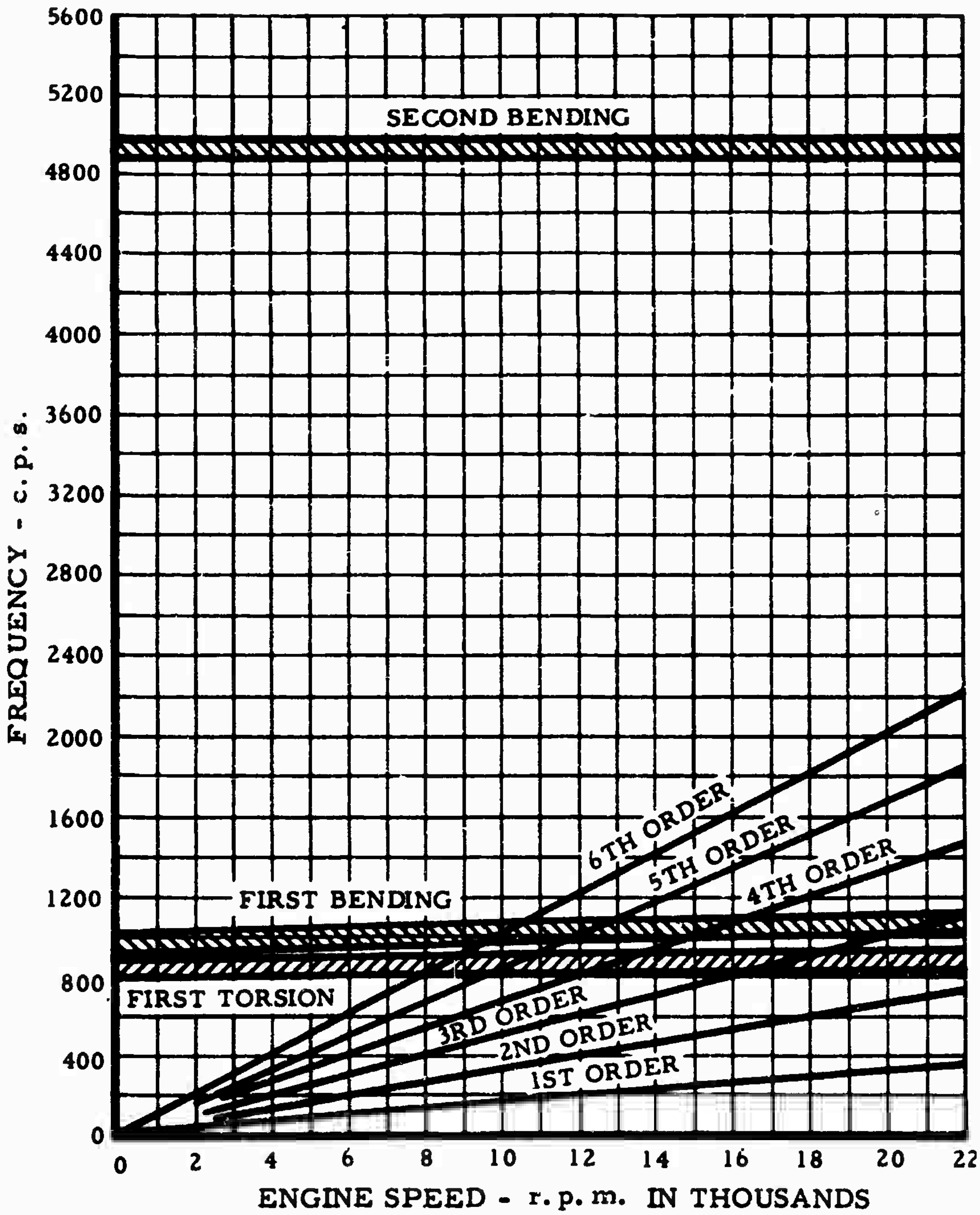


Figure 14. Inducer Blade Interference Diagram -- Continental Model 357-1 Engine.

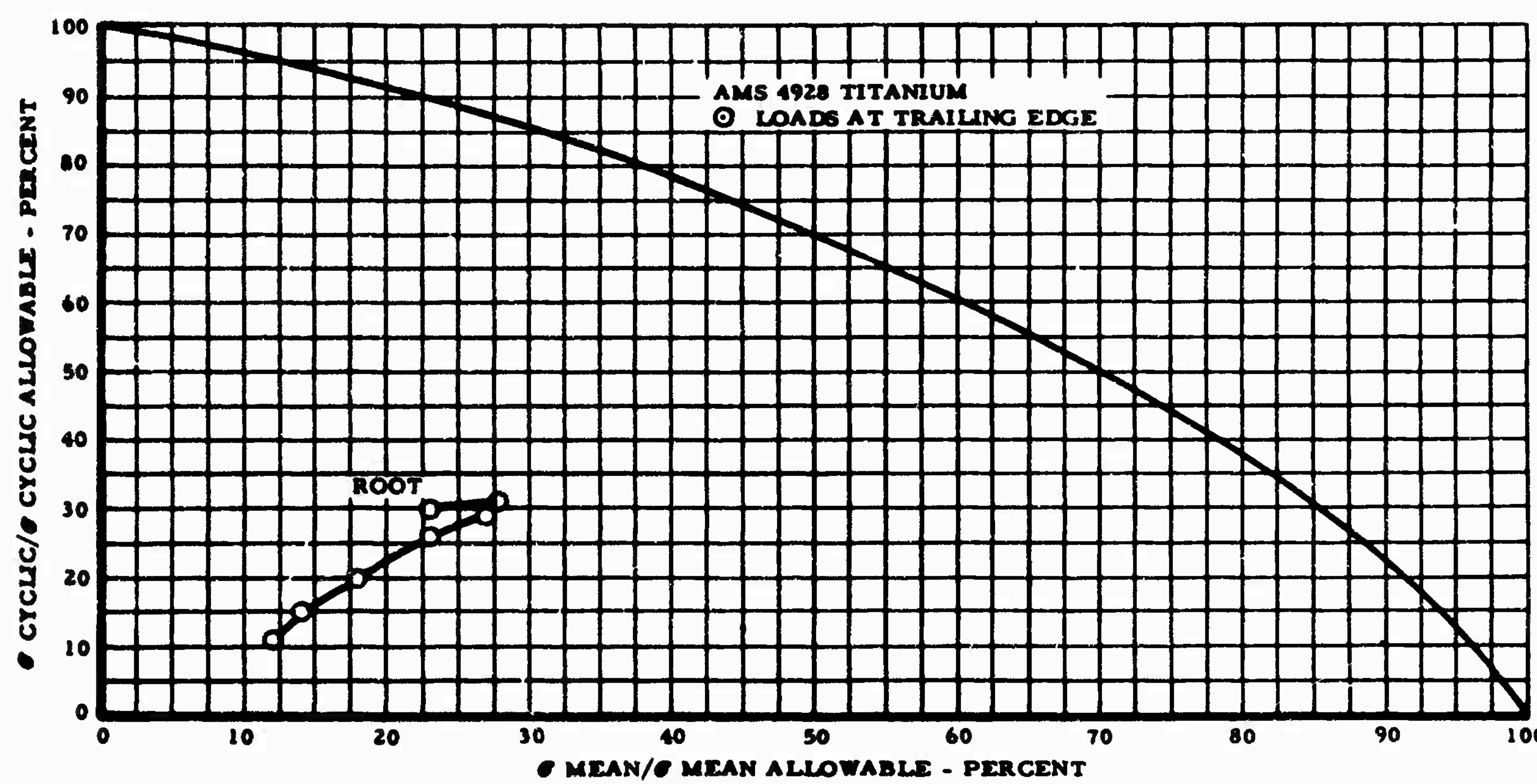


Figure 15. Modified Goodman Diagram for Continental Model 357-1 Inducer Blade.

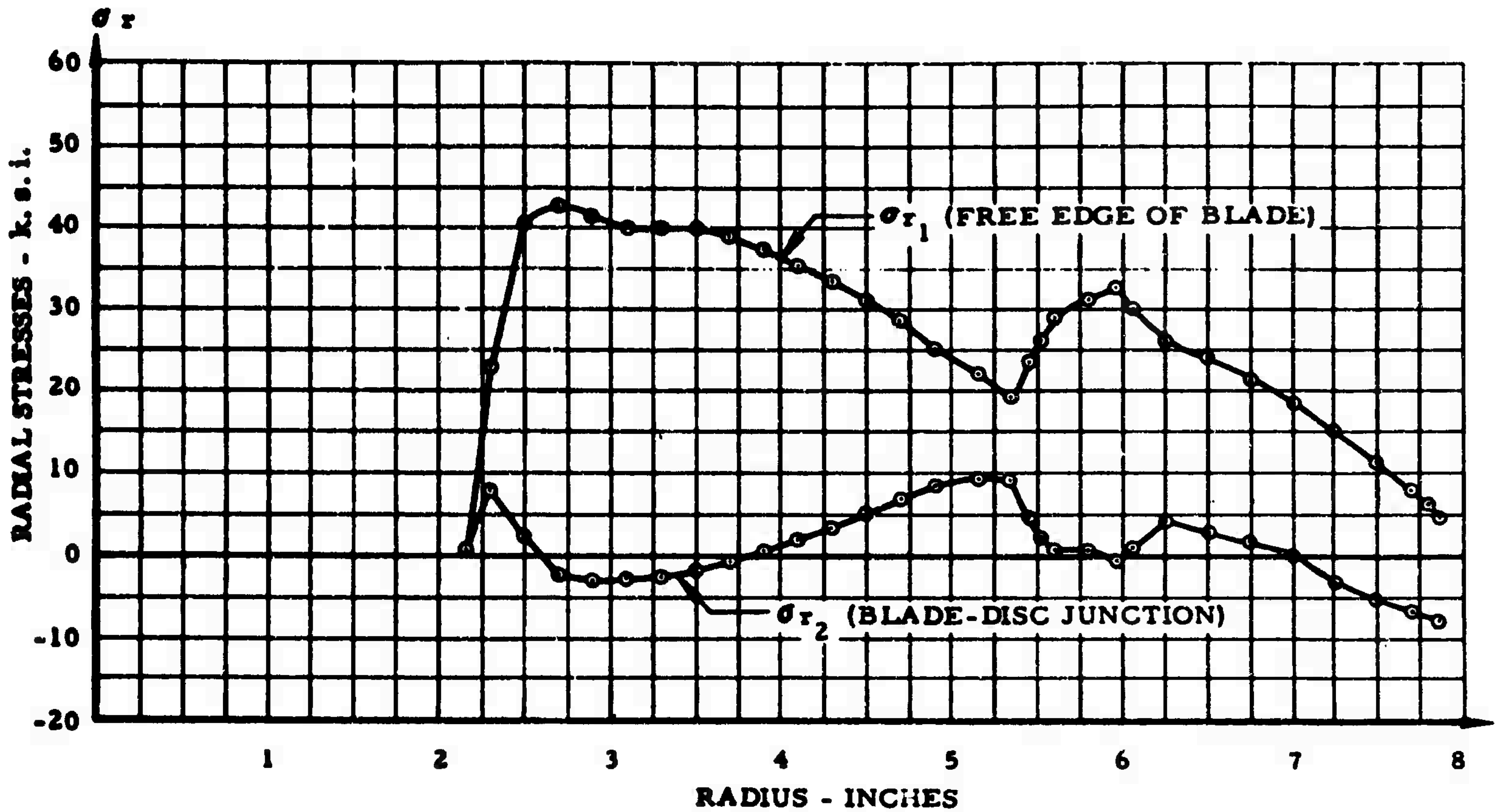


Figure 16. Radial Compressor Radial Blade Stress Summary - Continental Model 357-1 Engine.

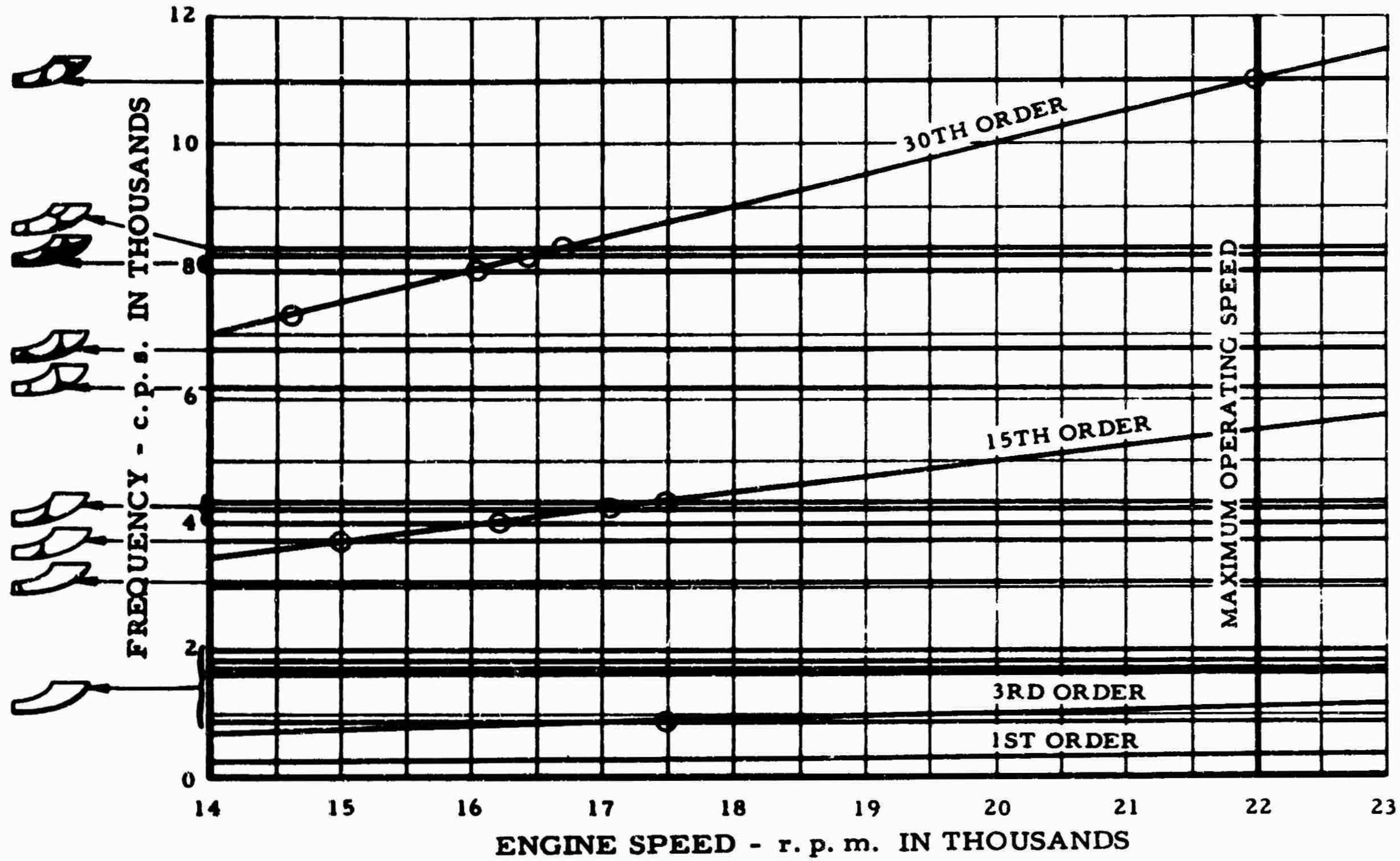


Figure 17. Radial Compressor Rotor Blade Interference Diagram - Continental Model 357-1 Engine.

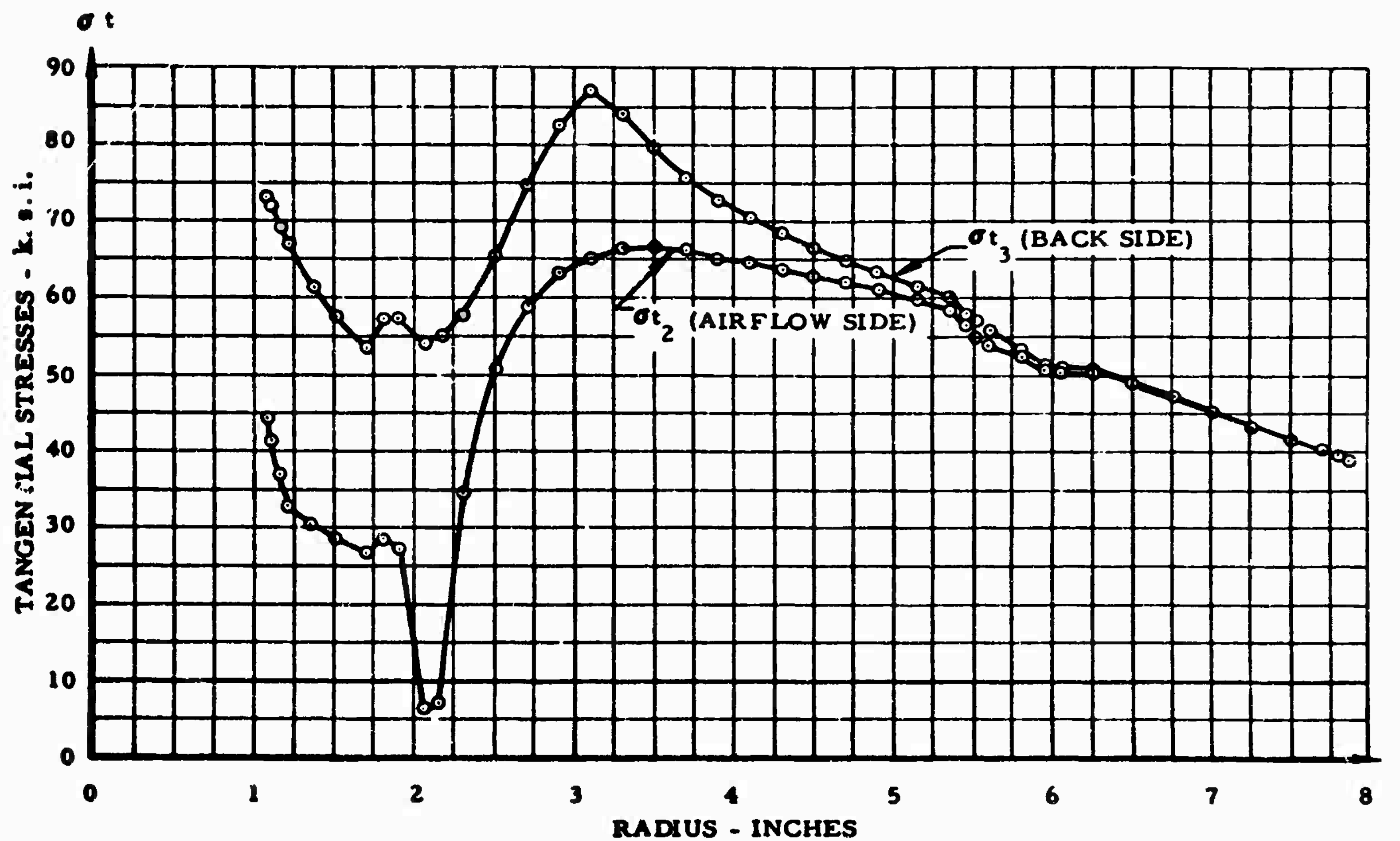


Figure 18. Radial Compressor Tangential Stresses - Continental Model 357-1 Engine.

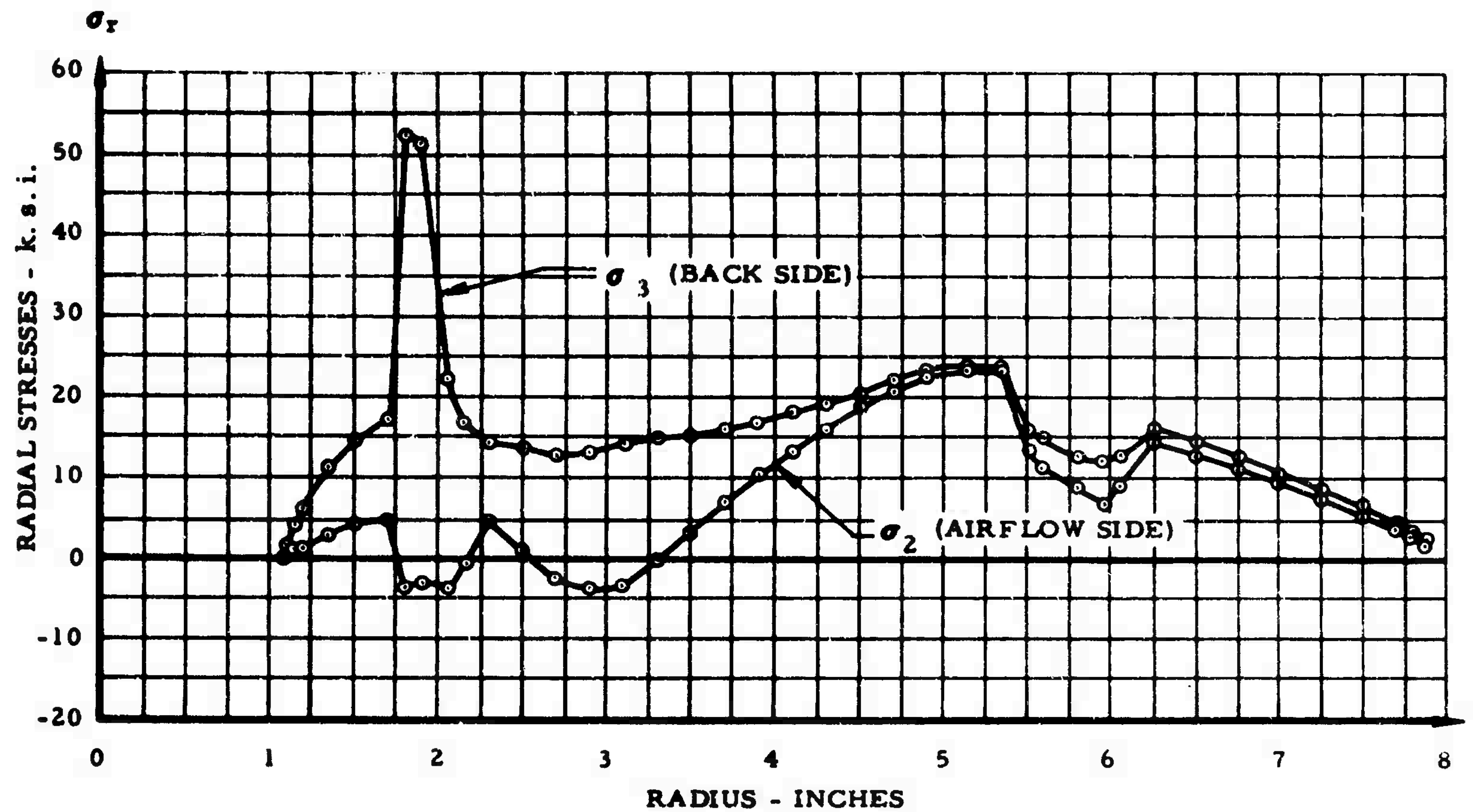


Figure 19. Radial Compressor Radial Disc Stresses - Continental Model 357-1 Engine.

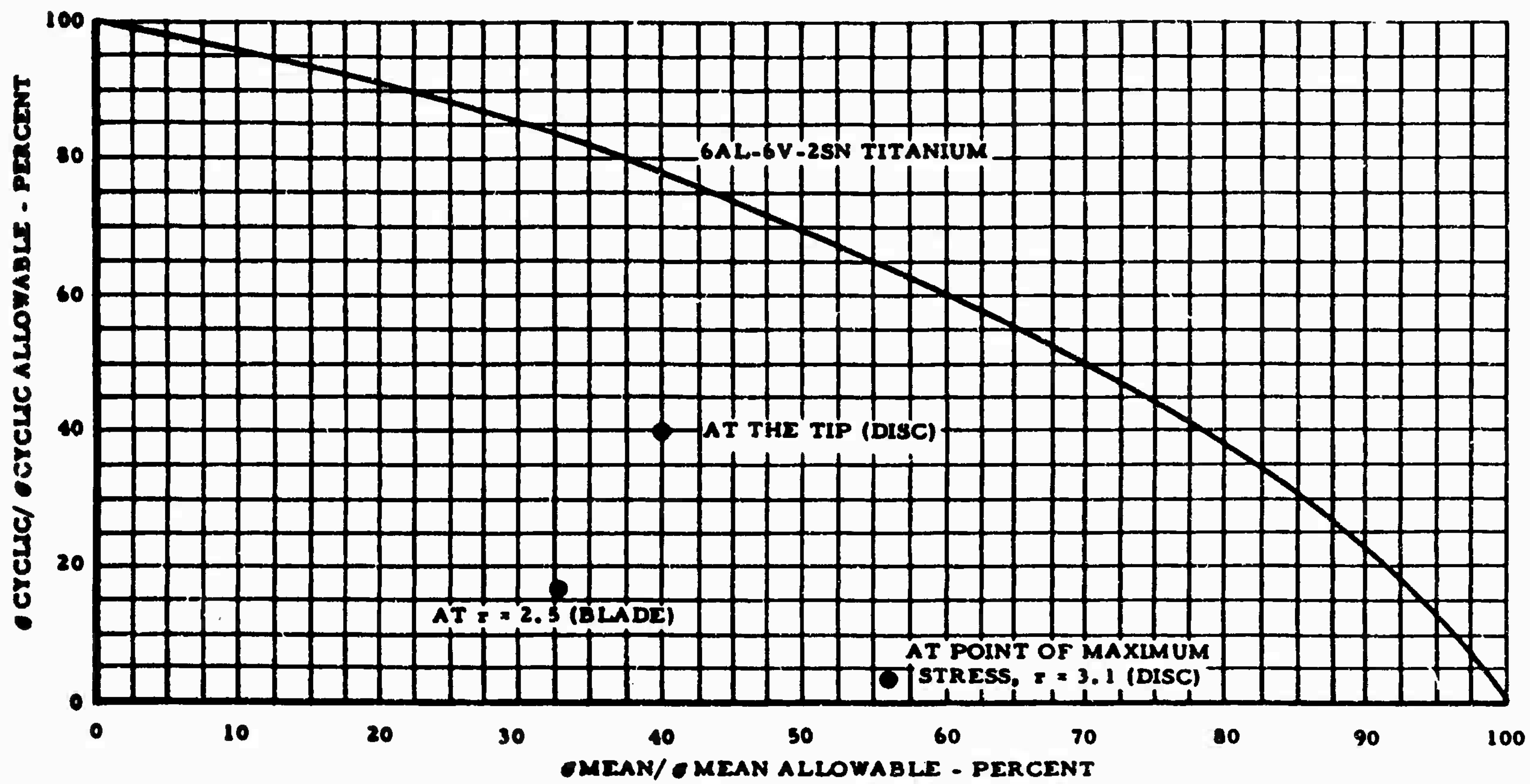


Figure 20. Modified Goodman Diagram for Continental Model 357-1 Radial Compressor Blade and Disc.

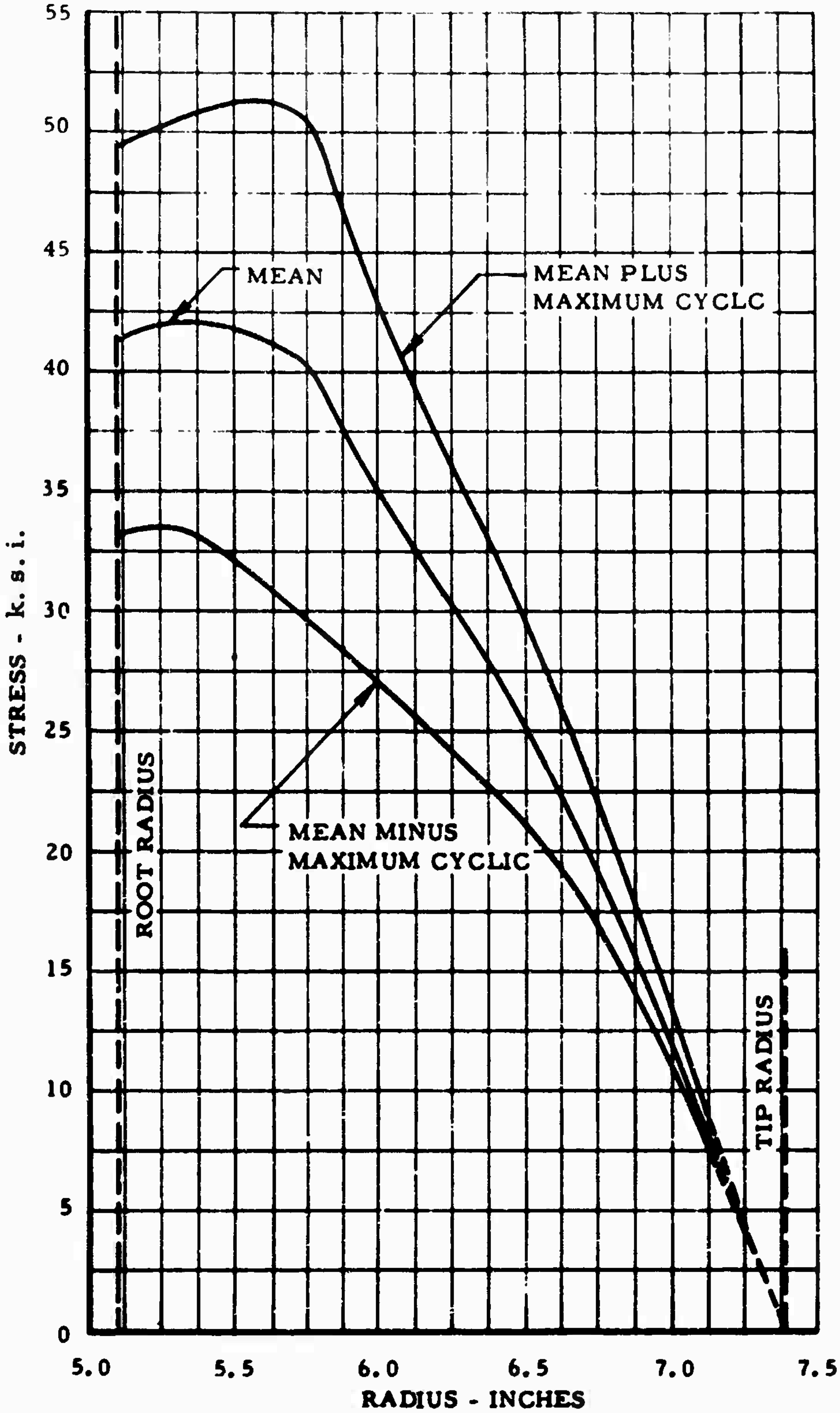


Figure 21. Turbine Blade Stress Summary - Continental Model 357-1 Engine.

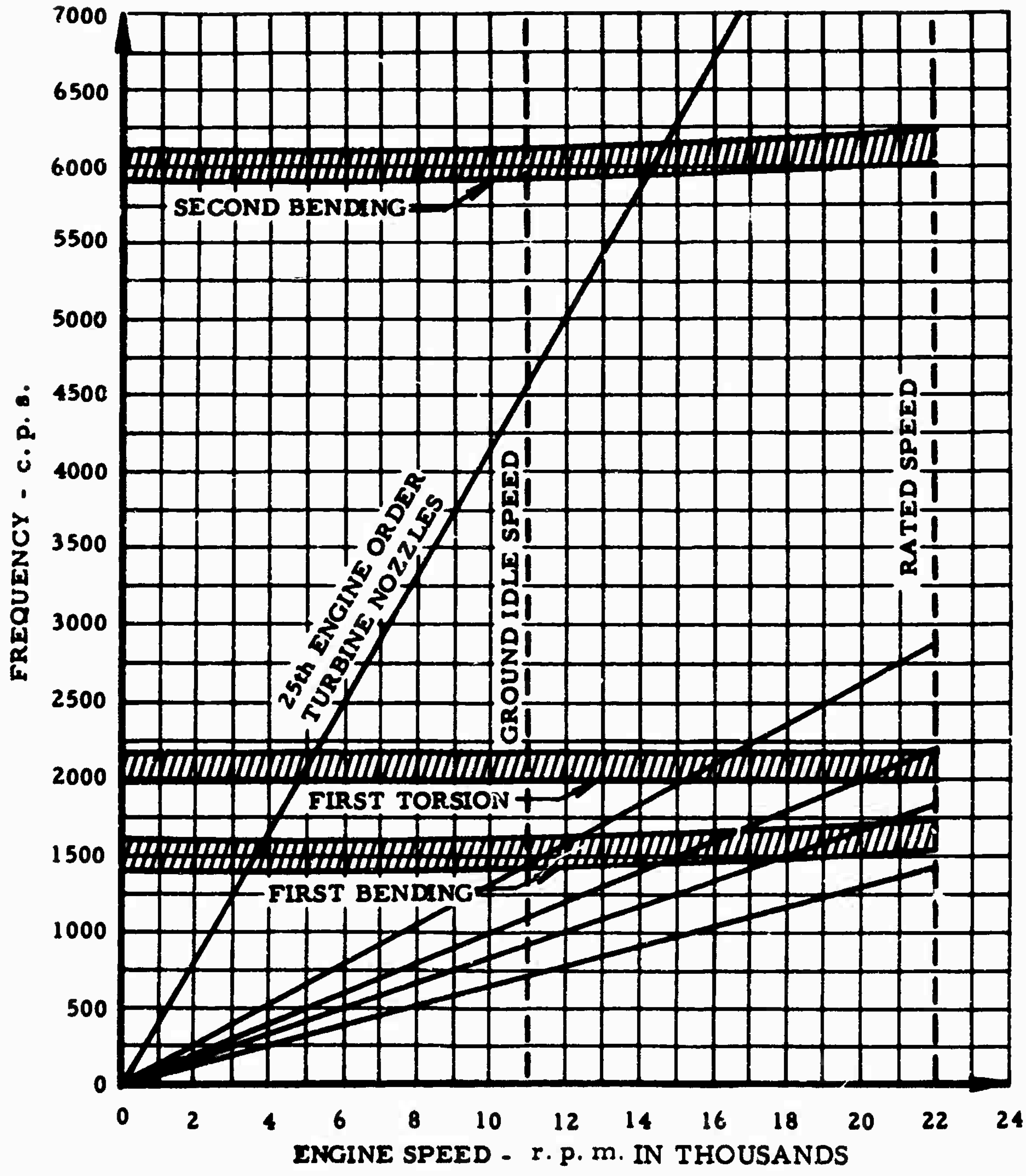


Figure 22. Turbine Blade Interference Diagram - Continental Model 357-1 Engine.

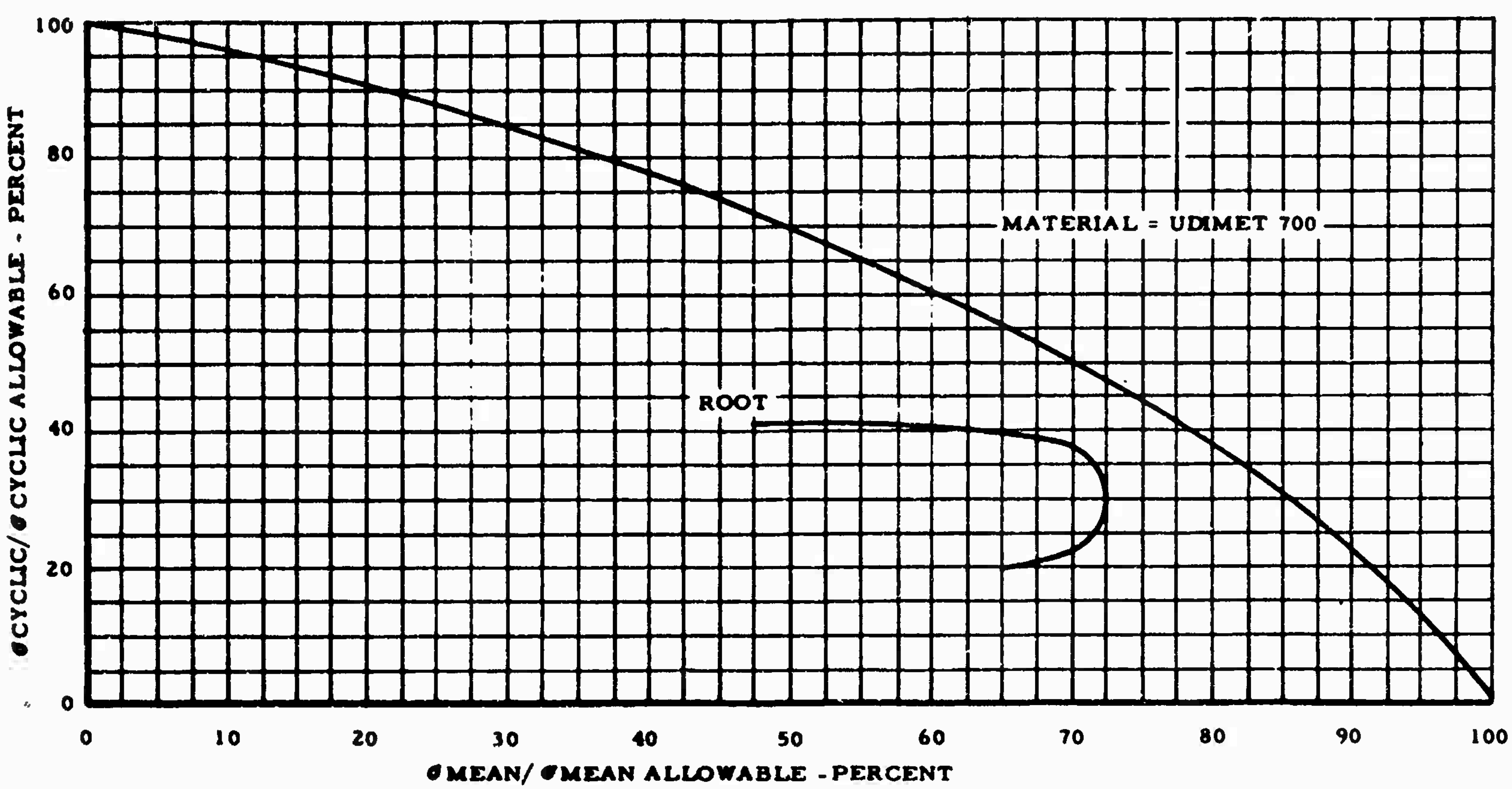


Figure 23. Modified Goodman Diagram for Continental Model 357-1 Turbine Blade.

stress rupture stress. The disc mean stress and the mechanical property summary for Udimet 700 material is shown in Figure 24. The modified Goodman diagram for the disc is shown in Figure 25.

The stress and vibration levels for the disc compared to the material properties show an adequate structural margin for the turbine.

CRITICAL SPEEDS AND SHAFT STRUCTURAL ANALYSIS

The principal source of mechanical vibration in a gas turbine engine is unbalance in the rotor system. This unbalance may excite critical speeds in the rotor itself or induce resonance in any combination of structural elements that may be tuned to the recurring frequency of the rotor. If sufficient flexibility exists in a rotor system, locked-in bending moments caused by the uneven distribution of unbalance along the length of the rotor may cause the shaft to deflect at high speeds. This deflection will destroy the balance that was previously obtained at lower balancing speeds.

The lightweight, high bending modulus rotor system, combined with the high spring rate rotor suspension system and the short spans between bearing supports, has resulted in a design that permits reasonable bearing loads and shaft deflection under the environmental conditions. This combination of rotor and supporting system compliances has also minimized the effect of rotor unbalance and critical speeds.

The shaft flexural critical speeds are above the engine operating range. These values are computed using finite spring rate at the bearing supports and giving full consideration to the gyroscopic stiffening effects of high mass moment of inertia components in the rotor system.

The critical speeds and the bearing support spring rates are given below.

Critical Speeds:

Compressor Shaft	=	41,000 r.p.m.
Turbine Shaft	=	38,300 r.p.m.

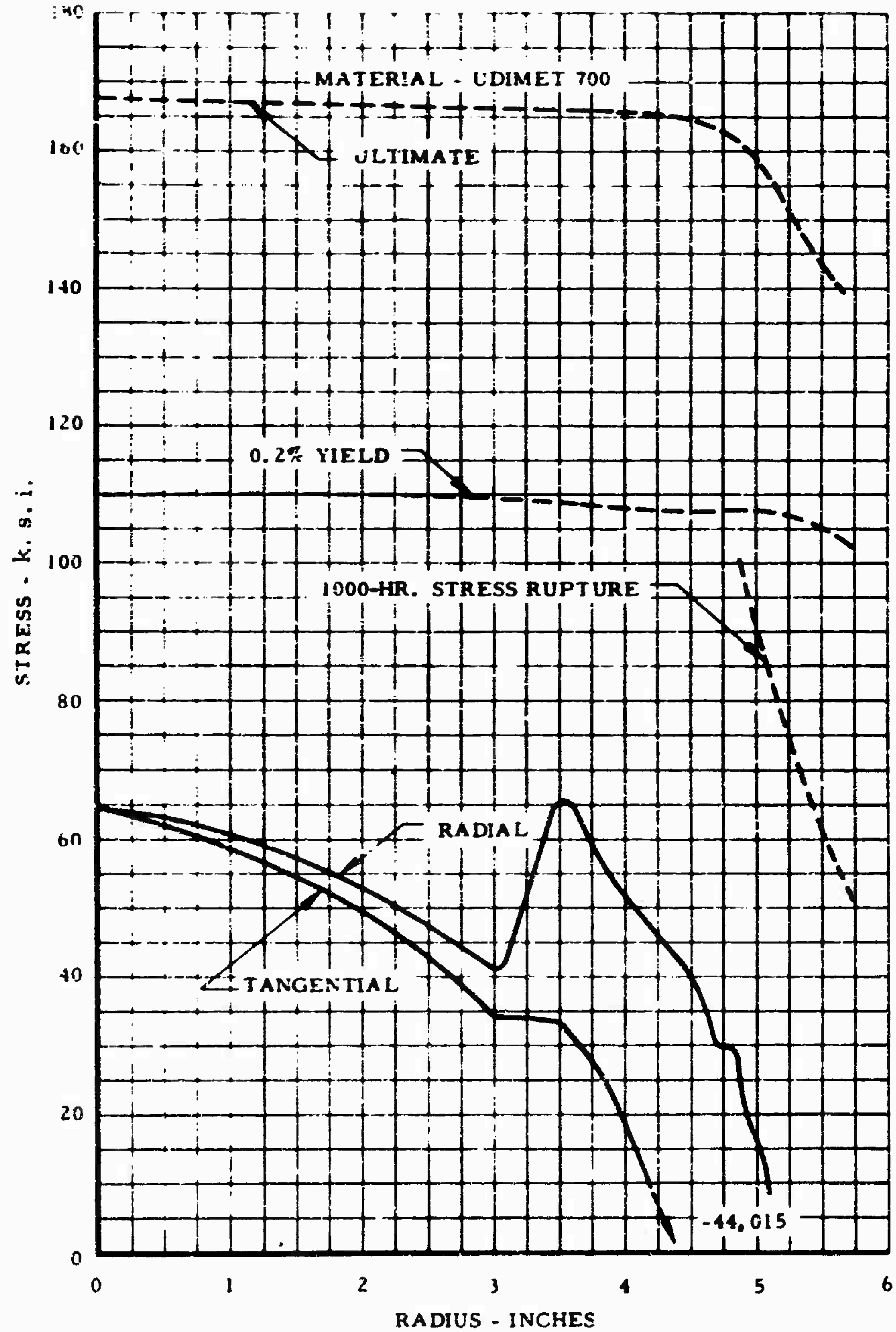


Figure 24. Turbine Disc Steady-State Stresses - Continental Model 357-1 Engine.

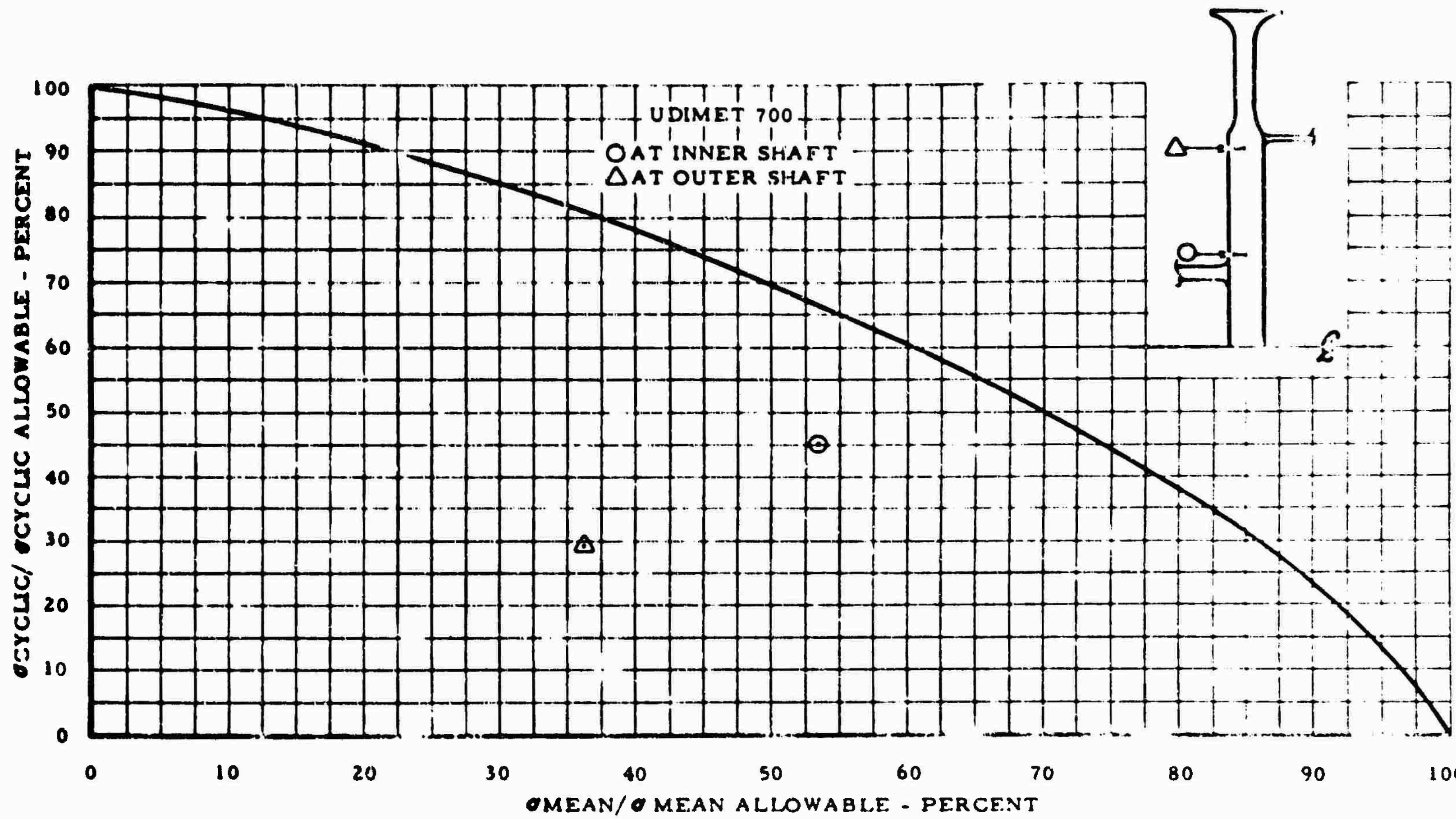


Figure 25. Modified Goodman Diagram for Continental Model 357-1 Turbine Disc.

Spring Rates:

No. 1 Bearing Support	=	5.67×10^5 lb/in.
No. 2 Bearing Support	=	10^8 lb/in.
No. 3 Bearing Support	=	10^8 lb/in.
No. 4 Bearing Support	=	6.52×10^5 lb/in.

The torsional critical speed of the turbine is 9,300 r.p.m., placing it slightly below the engine idle speed.

The maximum cyclic compressor shaft stress is 18,100 p.s.i. The endurance limit of the AMS 4928 titanium shaft material is 53,000 p.s.i., with adequate structural margin.

The maximum cyclic turbine shaft stress is 11,600 p.s.i. The endurance limit of the Waspaloy shaft material is 50,000 p.s.i., with adequate structural margin.

STATIC STRUCTURE

The application of the Model 357-1 engine has required that particular emphasis be placed upon the rigidity of the static structure. Maximum section moduli with minimum weight have been obtained by the use of annular structures without longitudinal joints. Minimum shear loads and bending moments have been effected through the use of lightweight materials and short length multipurpose components.

Components subject to temperature gradients have been designed with gradually varying sections to keep thermal stresses within the material capabilities. Flanges necessarily represent abrupt changes in section, and all have been isolated from the high temperature areas by suitable conical or cylindrical transition sections. These sections are designed to provide compliance between the relatively stiff flange and the thermally cycled hot parts. Maximum thermal stresses are experienced during temperature transients. Flange materials have been chosen that will provide yield points above the high transient stresses as well as suitable stress rupture properties at stabilized temperature levels.

Care has been taken to make welds only in areas of gradual section changes. Struts and bosses have extended lips tapering to approximately sheet thickness at the weld areas. Annular flanges have conical or cylindrical extensions gradually tapering to sheet thickness at the weld areas. These extensions provide essentially identical heat conduction sections at the weld in order to realize maximum physical properties in the weld regions. The welds, being in areas with gradual section change, are subjected to minimum thermal stresses.

The basic structural support of the Model 357-1 engine, Figure 26, is the center support ring (1). The ring directly supports the radial loads of the center bearings and the center support members. The front bearing support and the attached accessories are cantilevered from the center support ring by the compressor housing. Likewise, the rear bearing support and the exhaust nozzle are cantilevered by the combustor housing. The primary engine support lugs are an integral part of this center support ring.

The center support lugs, located 60 degrees apart on the inboard side of the engine, support the centrifugal force of the engine, the precession moment of the engine rotor, the vertical (flapping) acceleration forces, the shear forces resulting from the chordwise (lead and lag) acceleration of the helicopter rotor tip, and the shear force of the engine thrust.

An engine stabilizing lug is located on the inboard side of the rear bearing support ring (20). This lug is designed to take reactions in line with the helicopter rotor blade, allowing free motion in other directions. The basic loads supported by this lug are the reactions resulting from moments of the engine forces about the center support lug. These include the centrifugal force acting through the engine center of gravity normal to the helicopter rotor axis, and the chordwise acceleration forces and the engine thrust acting parallel with the engine rotor axis.

The front bearing support (2) is a one-piece aluminum casting consisting of the front support ring, five radial struts, and a front bearing housing. The support is provided with cored and drilled passages for the accessory drive shafts, pressure and scavenge oil, fuel supply, overboard drains, anti-icing, and labyrinth seal pressurization air.

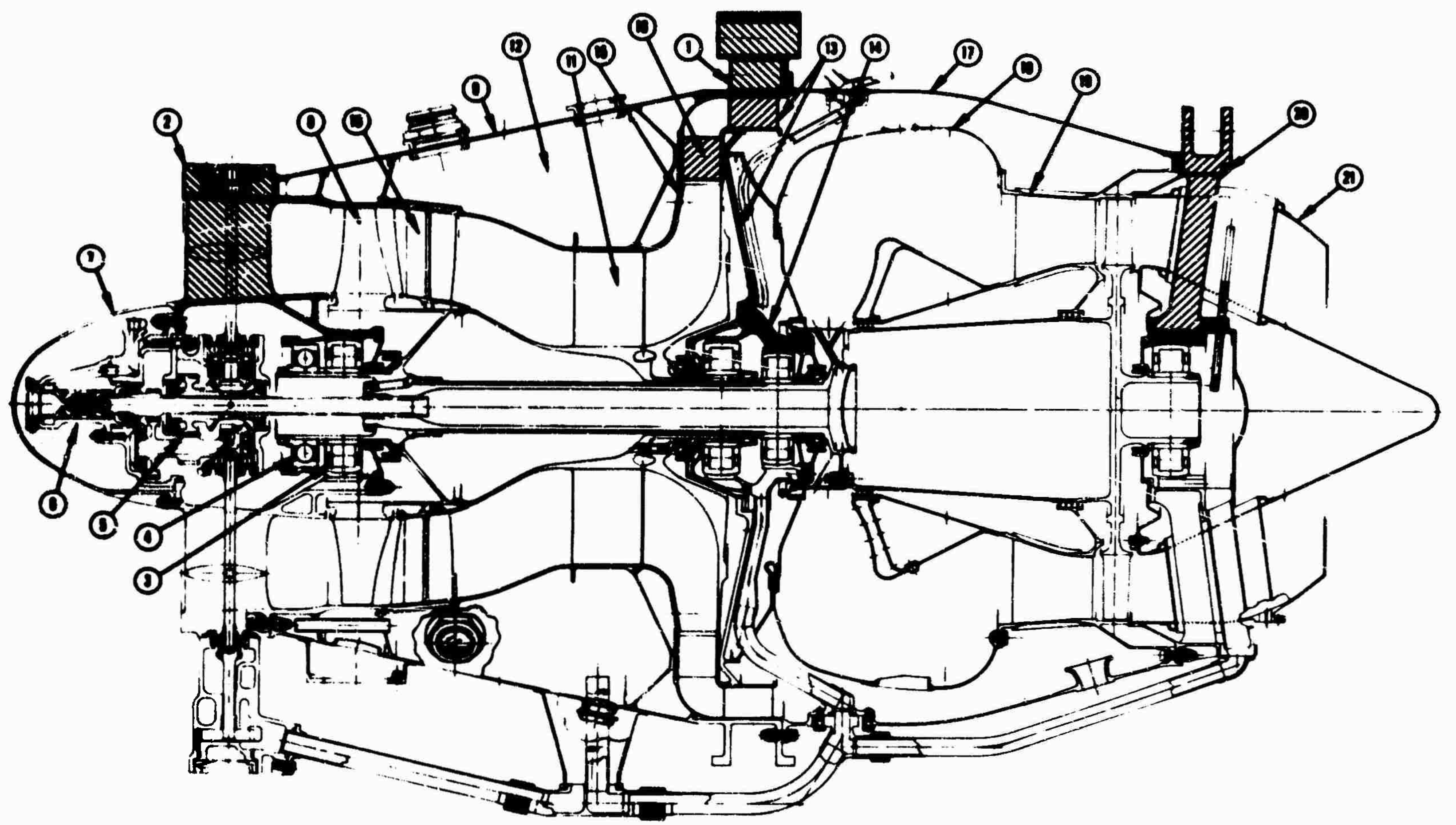


Figure 26. Continental Model 357-1 Engine Static Structure.

The front bearing housing supports the compressor shaft front radial bearing (3), the engine thrust bearing (4), the internal accessory gear drive (5), the fuel transfer labyrinth seal (6), and the inlet nose cone (7).

The front bearing housing and the front support ring are connected by five struts, crossing the airflow path. The boundary surfaces form a smooth contour for the air entry into the axial compressor (8). Three of the struts shroud the accessory drive shafts to the fuel pump, fuel control, and the oil pump. The orientation of these accessories in the centrifugal field has dictated the location of the struts. The remaining struts are spaced for load distribution and structural rigidity of the support.

The front support ring furnishes the mounting and accessory drive pads for the engine accessories. A cylindrical extension serves as the axial compressor shroud, while forming a closure with conical extensions of the compressor housing (9) to create the anti-icing and air start annuli. This extension also traps the axial stators (10), and prevents their rotation by a slot and tang engagement. The outer support ring is piloted and bolted to the front flange of the compressor housing.

The axial stators (10) are precision-cast aluminum to provide close dimensional and form control. They are split in two circumferential segments, doweled, and machined as an assembly. The stators are nonstructural elements, thus keeping distortion to a minimum and stress levels low.

The titanium compressor housing (9), is fabricated of an inner and outer shell joined at both ends. The outer shell is integral with the center support ring and forms the primary front structural casing. The inner shell forms the shroud for the radial inducer (11), the outer boundary of the inter-compressor air passage, and the pilot for the axial compressor stators. The cavity formed by the closure of the two shells serves as the engine oil tank (12).

The compressor housing is redundant with respect to the transmission of axial and radial loads. However, the structural members supporting the inner shell are axially flexible relative to the conical member transmitting the load to the outer shell. The major portion of the internal gas loads is transmitted through the inner shell, resulting in axial deflections which do not affect critical clearances.

A steel weldment consisting of the cast axial diffuser segments and the forged, fully-machined, center support diaphragm (13), is piloted in the center support ring and trapped by the combustor housing (17). The center bearing housing (14) is bolted to the inside diameter of the center support diaphragm. This cast and machined housing supports the radial loads of the compressor shaft rear bearing and the turbine shaft front bearing.

The radial compressor shroud (15) and the radial diffuser assembly (16) are trapped in the compressor housing by the center support diaphragm. These components are mutually interlocked, and the center support diaphragm assembly is prevented from rotating by a slot and tang engagement with the combustor housing. The differential pressure acting on these components is transmitted as an axial load through the radial diffuser and the radial compressor shroud to the conical segments of the compressor housing.

The combustor housing (17) is a steel casing with a welded flange at each end. The front flange is bolted to the center support ring. The rear flange is bolted to and supports the rear bearing support (20), exhaust nozzle (21), combustor shell (18), and turbine inlet nozzle assembly (19).

The rear bearing support (20) is a welded assembly of forged and machined steel. The support consists of an outer ring, five support struts, and an inner housing. The support houses the turbine shaft rear bearing and supports the exhaust nozzle through a conical isolation section. The turbine shaft rear bearing radial load and the axial gas loads exerted on the exhaust nozzle are transmitted from the rear bearing support through the combustor housing to the center support flange.

The exhaust nozzle (21) is a sheet metal weldment consisting of the outer duct, five vanes, and the inner cone. The inner cone is supported from the outer duct by the vanes. The vanes also shield the support struts, oil supply and scavenge lines, and the labyrinth seal air supply from the turbine exhaust gases.

The exhaust nozzle is required to support only its own weight in addition to a small axial load. However, deformation of the nozzle in the g field will critically affect engine performance. The plug-type nozzle has been designed for low shear loads and bending moments at the support structure with high conical rigidity at the critical flow areas.

The turbine inlet nozzle (19) is a welded assembly consisting of an outer ring, 25 hollow vanes, and an inner cone. The assembly forms the boundary passage for the combustion gases, directing the flow into the turbine. The assembly is enclosed by relatively cool compressor discharge air, which passes through the hollow passages of the vanes and through the inner combustor shell assembly to the combustion chamber.

The inner cone supports the inner combustor shell assembly and serves as labyrinth seal shrouds at the rear of the fuel slinger and at the turbine inlet. The inner combustor shell consists of a conical support and a perforated dish which directs air into the primary combustion zone and absorbs thermal radiation. The two parts are welded at their inner diameters and are free to expand in radial and axial directions.

The outer ring of the turbine inlet nozzle has a cylindrical extension which serves as the turbine shroud. The ring is flanged at each end. The front flange pilots and is bolted to the outer combustor assembly.

The outer combustor (18) is fabricated in two pieces: the outer shell and the swirl vane. The swirl vane, which directs primary airflow into the combustion chamber, is located axially to the outer combustor shell by spring clips. The vane is piloted at the inner diameter and restrained from axial deformation by the center bearing housing rear labyrinth seal. This construction allows free radial expansion of the vane, which is subject to large thermal gradients. A conical support is welded to the front of the outer combustor shell. This support is perforated and slotted to provide passages for the primary combustor air and the center bearing housing oil supply and scavenge lines.

The bolted assembly, consisting of the outer combustor shell and the turbine inlet nozzle, is attached to the combustor housing rear flange through a conical transition segment. This assembly is radially supported at the front end by the outer combustor shell support piloted by a cylindrical extension of the center support diaphragm (13). Therefore, the assembly is radially supported at both ends and is free to expand axially in the forward direction.

The radial support required for the outer combustor shell and turbine inlet nozzle assembly is for its own distributed load in the g field. The pressure differential across the turbine inlet nozzles and the turbine inlet labyrinth seal results in an axial load that is transmitted through the combustor housing to the center support ring.

The general structural philosophy has been to use large-diameter, continuous structure with suitably compliant sections that permit tightly bolted joints. This eliminates the necessity for expensive sliding joints, which are subject to fretting, high frequency vibration, and higher manufacturing costs.

Front Bearing Support Frame

The front bearing support frame is a cast aluminum ring and spoke structure. The stress and spring rate summary is shown in Figure 27. The material yield point is 33,000 p.s.i. and the ultimate is 43,000 p.s.i.

The maximum stresses are below the yield in both the 235 and 259g condition. Localized yielding may occur at the 367g overspeed condition but this is considered satisfactory.

Center Support Ring

The center support ring is machined from a titanium forging. The ring is stiffened internally by the center main bearing diaphragm.

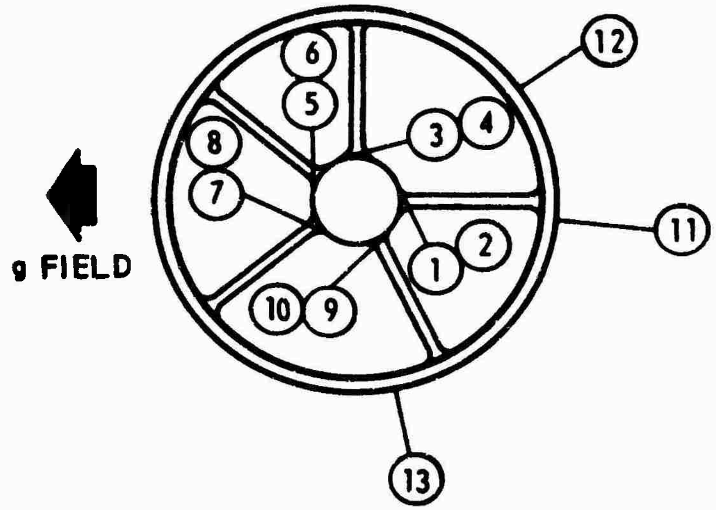
The support ring stresses are essentially symmetrical above and below the horizontal center plane of the engine. The highest stresses exist in the 0- to 90- degree quadrant, and these values are tabulated in Figure 28. These values are shown for the 235, 259, and 367g conditions.

The ring material is 6AL-6V-2Sn titanium with an ultimate of 150,000 p.s.i. and a yield of 140,000 p.s.i.

The ring stresses are well below the yield for the 235 and 259 g condition. Localized yielding may occur at the 367g overspeed condition but this is considered satisfactory.

Center Bearing Support (Compressor Cover)

The center bearing support is fabricated from cast steel vanes along the OD and a fully machined steel forging in the center.

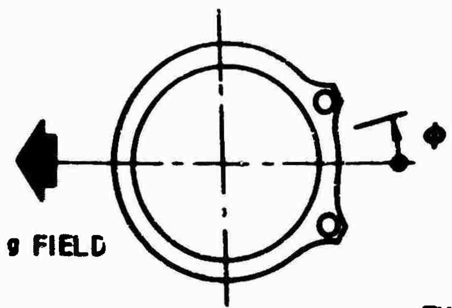


FRONT BEARING SUPPORT FRAME

SPRING RATES:
MINIMUM AXIAL - 6.34×10^6 lb/in
MINIMUM RADIAL - 5.67×10^5 lb/in

	LOCATION	TYPE	STRESS IN p.s.i. 235g	259g	367g
1.	STRUT L.E.	BEND. + P/A	-15,080	-16,270	-21,600
2.	STRUT T.E.	BEND. + P/A	27,210	29,660	40,670
3.	STRUT L.E.	BEND. + P/A	-11,060	-11,840	-15,330
4.	STRUT T.E.	BEND. + P/A	16,810	18,190	24,420
5.	STRUT L.E.	BEND. + P/A	- 3,540	- 3,700	- 4,450
6.	STRUT T.E.	BEND. + P/A	-21,790	-24,120	-34,620
7.	STRUT L.E.	BEND. + P/A	- 9,370	10,690	16,600
8.	STRUT T.E.	BEND. + P/A	-23,090	-25,780	-37,890
9.	STRUT L.E.	BEND. + P/A	- 4,190	- 4,425	- 5,480
10.	STRUT T.E.	BEND. + P/A	- 1,100	- 1,337	- 2,350
11.	RING O.D.	BENDING	-22,230	-24,500	-34,720
12.	RING O.D.	BENDING	21,870	24,100	34,150
13.	RING O.D.	BENDING	20,690	22,800	32,300

Figure 27. Continental Model 357-1 Engine Front Bearing Support Frame Stress Summary.

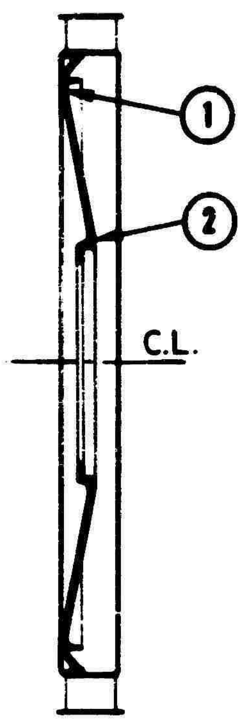


ENGINE SUPPORT RING

Φ DEG	g FIELD	BENDING STRESS k.s.i.	HOOP STRESS k.s.i.	COMBINED STRESS k.s.i.
0	235	51.2	39.5	90.7
	259	56.4	43.5	99.9
	367	80.0	60.1	140.1
10	235	33.9	37.6	71.5
	259	37.4	41.4	78.8
	367	53.0	58.8	111.8
20	235	12.8	35.0	47.8
	259	14.1	38.6	52.7
	367	20.0	54.6	74.6
30	235	22.7	19.1	41.8
	259	25.0	21.1	46.1
	367	35.4	29.9	65.3
40	235	9.6	49.3	58.9
	359	10.6	54.3	64.9
	367	15.0	77.0	92.0
50	235	14.7	48.9	63.6
	259	16.3	53.9	70.2
	367	23.0	76.3	99.3
60	235	27.6	46.1	73.7
	259	30.4	50.8	81.2
	367	43.0	72.1	115.1
65	235	28.8	44.8	73.6
	259	31.7	49.4	81.1
	367	45.0	70.0	115.0
70	235	26.9	43.9	70.8
	259	29.6	48.4	78.0
	367	42.0	68.6	110.6
80	235	16.6	40.1	56.7
	259	18.3	44.2	62.5
	367	26.0	62.6	88.6
90	235	5.1	38.3	43.4
	259	5.7	42.2	47.9
	367	8.1	59.8	67.9

Figure 28. Continental Model 357-1 Engine Support Ring Stresses.

The material is 17-4 PH stainless steel with a yield of 170,000 p.s.i. and an ultimate of 190,000 p.s.i. The maximum stresses presented in Figure 29 show a substantial structural margin. These stresses are computed from a combination of the air pressure load and the g field load.



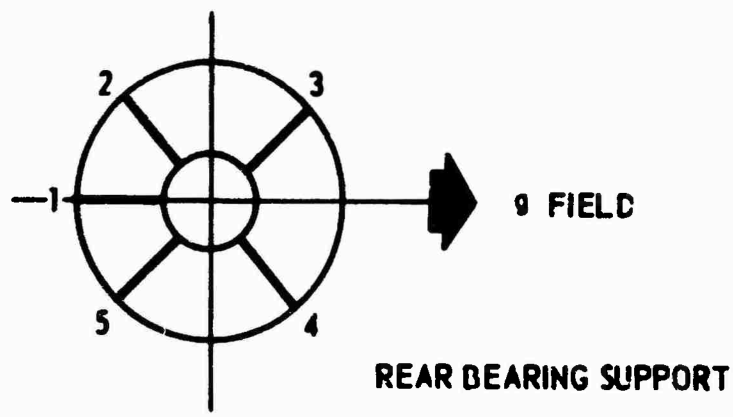
POINT	STRESS DIRECTION	STRESS k.s.i. 259g	STRESS k.s.i. 367g
1	TANGENTIAL	41.[illegible]	56.6
	RADIAL	38.8	54.9
2	TANGENTIAL	60.6	67.0
	RADIAL	1.2	11.2

Figure 29. Continental Model 357-1 Compressor Cover Stresses.

Rear Bearing Support

The rear bearing support is a welded spoke and ring structure. The material is Inco 718 nickel alloy with 127,000 p.s.i. yield and 147,000 p.s.i. ultimate at 1,000°F. The stresses presented in Figure 30 show an adequate structural margin.

A summary of the Model 357-1 structural steel housing stresses is presented in Table 3. Table 4 summarizes the combustor and turbine inlet nozzle stresses at 259g.



STRUT	THERMAL STRESS k.s.i.	THERMAL STRESS PLUS 235g STRESS k.s.i.	THERMAL STRESS PLUS 259g STRESS k.s.i.	THERMAL STRESS PLUS 367g STRESS k.s.i.
1	13.3	20.2	21.0	25.3
2	19.1	20.3	20.6	22.3
3	28.5	20.0	19.3	14.9
4	28.5	25.2	24.7	21.7
5	19.1	25.4	26.0	29.7

Figure 30. Continental Model 357-1 Rear Bearing Support Stresses.

TABLE 3
STRESS SUMMARY FOR STRUCTURAL SHEET HOUSINGS

Item	Longitudinal Stress, k.s.i. g Field	Tension	Comp.	Material Properties, k.s.i. Ultimate	Yield	Buckling
Compressor Housing*	259	23.7	14.1	150	140	23.7
	367	29.6	20.0			
Combustor Housing	259	22.6	17.9	147 at 600°F	98 at 600°F	35.3 at 600°F
	367	28.4	23.7			

* Hoop stress, 12,100 p.s.i.

The compressor housing and the combustor housing are large diameter, high inertia, structural beams. The relatively thin sheet metal construction produces a beam critical in buckling. The stress summary indicates considerable tensile yield strength margin and compressive stresses below the material buckling limit. The combustor housing physical properties are given at the maximum expected temperature for the parts.

TABLE 4
COMBUSTOR AND INLET NOZZLE (TIN)
STRESS SUMMARY AT 259g

Item	Stress (k. s. i.)	Type	Material Yield (k. s. i.)	Operating Temp. (°F)
Combustor Flange	29.5	Bending	35.0	1200
Combustor Shell	12.4	Bending	35.0	1200
Combustor Shell	11.0	Tangential	35.0	1200
TIN Inner Shell	5.7	Shear	21.0	1000
TIN Vane	3.3	Shear	21.0	1200

The combustor structure is a large-diameter shell supported laterally at its forward and aft sections. In order to withstand the high g loads, the forward support was incorporated..

The stress summary and material properties indicate an adequate margin for the combustor and turbine inlet nozzle in the 259g field. The 367g stresses are not presented due to the fact that the combustor and turbine inlet nozzle would be completely contained within the external supporting structure in the event of their failure.

ACCESSORY DRIVE

The accessory drive train, Figure 31, consists of four spur gears and four bevel gears, which provide the three drives tabulated below:

Drive	Drive Horsepower
Oil Pump	2.5
Fuel Pump	8.0
Fuel Control	0.25
Total	10.75

The r.p.m. signal will be provided from the fuel control.

Referring to Figure 31, a high speed spur (1) originates from the center of the engine, drives a compound spur-bevel gear (3) through a compound idler (2). The bevel gear (3) drives three radially located bevel gears (5), (6), and (7). The three gears drive through three radial tower shafts (8), (9), and (10) to furnish the three accessory drives.

Operation in the tip turbojet environment requires the particular orientation of accessories shown. The fuel control axis must be horizontal, the fuel pump axis vertical, and the oil pump in the lower outboard quadrant.

The three accessory drive pads are located on the outside diameter at the front frame of the engine and are designed to suit the engine requirements. Accessory drive capabilities are tabulated below:

Drive	Gear Ratio	Torque Rating (lb-in.) Continuous	Static	Rotation*
Oil Pump	0.2984	24.2	-	Counterclockwise
Fuel Pump	0.2984	77.6	310	Counterclockwise
Fuel Control	0.2984	2.4	25	Counterclockwise

* Rotation is expressed viewing the mounting pad.

The gear teeth have been analyzed and designed against the three general types of failure: pitting, bending or tooth breakage, and scoring. The first two are essentially fatigue-type in character and are functions of the total number of stress cycles as well as magnitude of dynamic load. Scoring is usually caused by local friction welding and will occur on the first application of maximum speed and torque.

The design criteria used to evaluate the gears against each of the characteristic types of failure are: Hertz stress for pitting, root beam stress for bending, and PVT factor for scoring.

The load used in computing Hertz and beam stress is composed of the total transmitted load plus a dynamic increment load. This dynamic increment results from inertia forces developed in the gear train by instantaneous accelerations caused by inaccuracies in spacing,

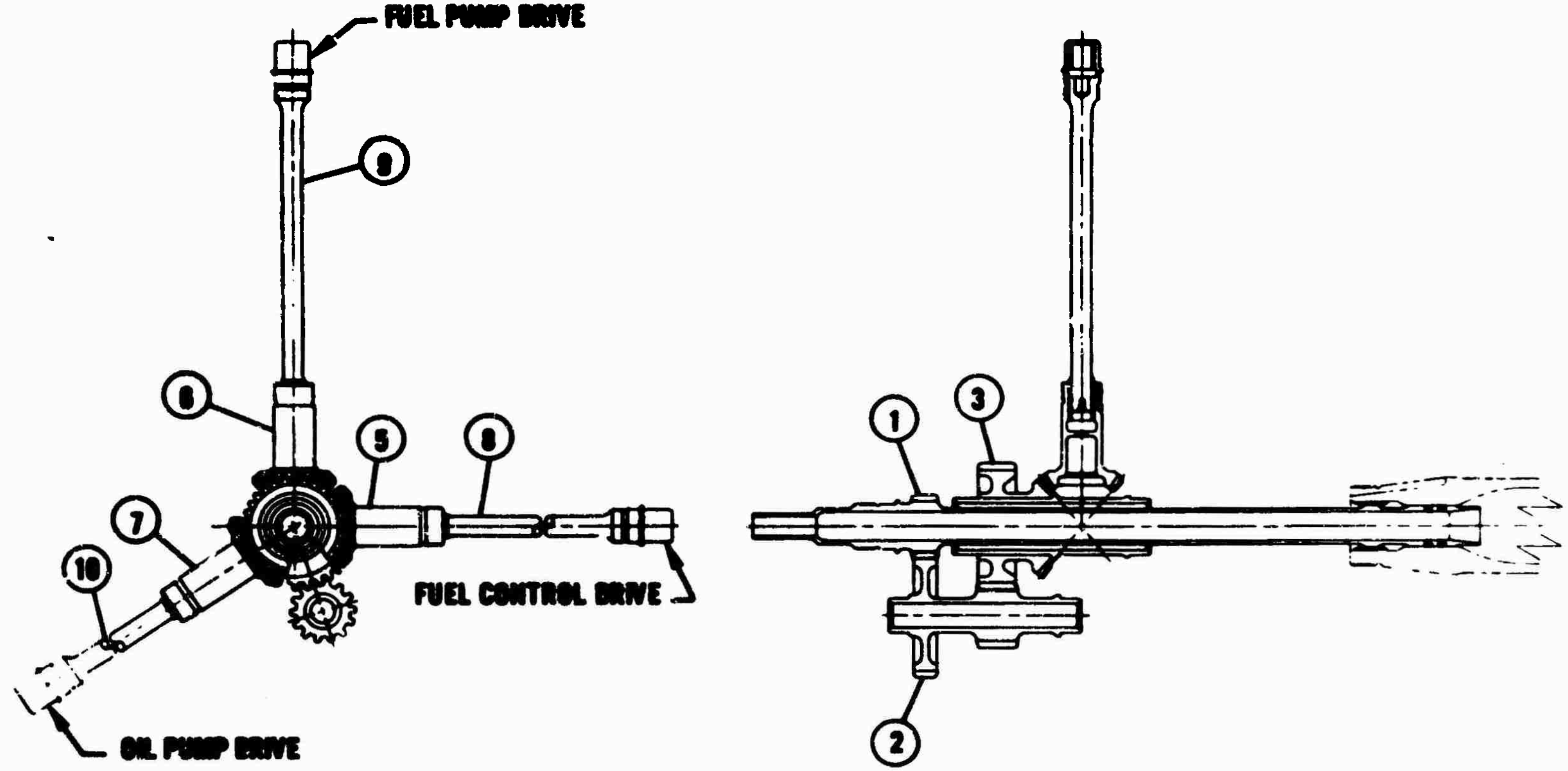


Figure 31. Continental Model 357-1 Accessory Drive Train.

thickness, and tooth profiles. A perfect involute will transmit uniform angular velocity, but any deviation from this form will cause a change in velocity which, over a very short time interval, represents a high acceleration of the gear mass and all other rigidly attached inertias in the train.

The method used in calculating the dynamic increment is based on Buckingham's works. Although various sources have indicated that the dynamic load as calculated by Buckingham's equation is larger than the actual dynamic load encountered, most authorities agree that it is the best method for comparing different designs. Continental's experience indicates that aircraft-quality, hardened and ground gears can operate up to 230,000 p.s.i. wear (Hertz) stress with little or no development.

Tooth bending stresses are calculated by applying the dynamic tooth load to the Lewis beam stress equation. This formula determines the equivalent stress that would exist on a constant strength parabola inscribed within the tooth, tangent at the base, and with its apex just intersecting the line of action of the applied tooth load. The stress limit that Continental has successfully permitted on previous designs is 100,000 p.s.i.

The static bending stress is computed as above but using only the static tooth load at the static torque rating.

The PVT scoring factor is a measure of the instantaneous rate of heat generation. It combines the Hertz contact pressure and the sliding velocity at the tip of the tooth with the length of the line of action from the pitch point to the tooth tip.

The safe limit for the scoring factor using very accurate case-hardened gears lubricated with a medium weight petroleum oil is 1,500,000 (dimensionless). Continental's experience with MIL-L-7808 turbine oil indicates that no decrease in scoring limit is necessary. The gears in this train have all been designed so that no PVT factor exceeds one-third the permissible safe unit.

The accessory gear stresses are presented in Table 5. Accessory bearing loads and B-10 lives are presented in Table 6.

Figure 32 is a schematic diagram of bearing location for the Model 357-1 engine.

TABLE 5 MODEL 357-1 ACCESSORY GEAR STRESSES			
Mesh	Wear Stress (p. s. i.)	Bending Stress (p. s. i.)	Static Bending Stress
High Speed Spur	226, 100	65, 500	33, 000
Low Speed Spur	223, 000	63, 800	40, 700
Oil Pump Bevel	159, 000	44, 000	6, 800
Fuel Pump Bevel	203, 000	71, 600	87, 400
Fuel Control Bevel	141, 000	34, 600	700

TABLE 6 MODEL 357-1 BEARING LOADS AND B-10 LIVES					
No. (Ref. Fig. 32)	Speed (r. p. m.)	Radial Load (lb.)	Thrust Load (lb.)	Equiv. Load (lb.)	Life, (Hours*)
1	22, 000	41. 3	0	41. 3	Over 10^4
2	10, 645	63. 2	0	63. 2	4850
3	10, 645	4. 3	0	4. 3	Over 10^4
4	5, 207	198. 0	49. 7	198. 0	1280
5	5, 207	118. 0	0	118. 0	6207
6	6, 565	6. 9	0	6. 9	Over 10^4
7	6, 565	2. 7	28. 4	60. 6	Over 10^4
8	6, 565	0	49. 4	76. 5	8820
9	6, 565	76. 0	0	76. 0	6500
10	6, 565	6. 0	32. 3	68. 6	Over 10^4
11	6, 565	7. 7	32. 0	54. 9	6360
12	6, 565	7. 7	0	7. 7	Over 10^4
13	6, 565	205. 0	0	205. 0	8290
14	6, 565	121. 0	29. 5	131. 2	8470
15	6, 565	24. 7	0	24. 7	Over 10^4

* All accessory drive bearing lives are based on continuous 100 percent speed and load.

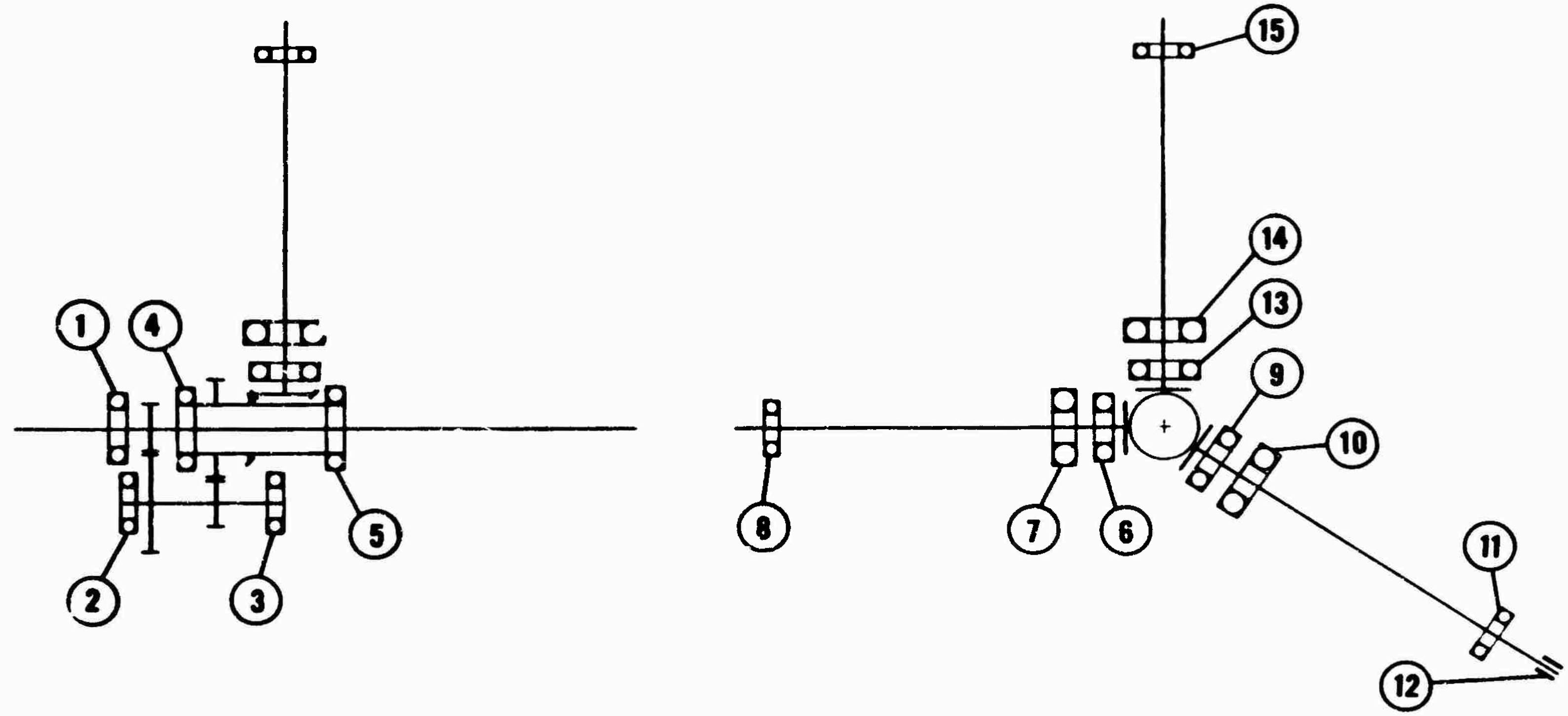


Figure 32. Continental Model 357-1 Bearing Locations.

SEALS

Labyrinth seals are used exclusively in the Model 357-1 engine because:

1. They are not subject to the wear problems that conventional face and lip seals would experience in the high g field.

2. They are considerably less sensitive to deflection.

3. They are virtually maintenance free.

4. They provide a far simpler engine assembly.

Straight, constant diameter, labyrinth seals have been used throughout the engine for simplicity, economy, ease of assembly, and their immunity to axial tolerances and thermal growth.

The seal element proportionality and overall configuration is based on previous engine experience as well as published literature. The seal configurations used have been individually idealized for sealing requirements, space utilization, radial and axial thermal growth, and to control the resultant rotor thrust load.

Lands of labyrinth seals of the rotating shaft structure are integral with the shaft to preclude the possibility of severe shaft damage in case of severe rub.

All the engine oil seals are designed with a low pressure differential and in such a manner that air flows into the oil compartments. The front and center bearing oil compartment balance pressure, Figure 33, is provided by holes in the compressor shaft that expose the air side of the oil seal to the axial compressor discharge pressure. The air passages are large relative to the expected airflows to assu the same pressure gradient across each oil seal.

Throttled compressor discharge air is provided between the seal lands at the rear bearing oil compartment, Figure 34. The airflow through the labyrinth seal along the back face of the turbine serves to purge the hot exhaust gas from the oil compartment. The balance of the air provides a relatively cool airflow into the rear bearing oil compartment.

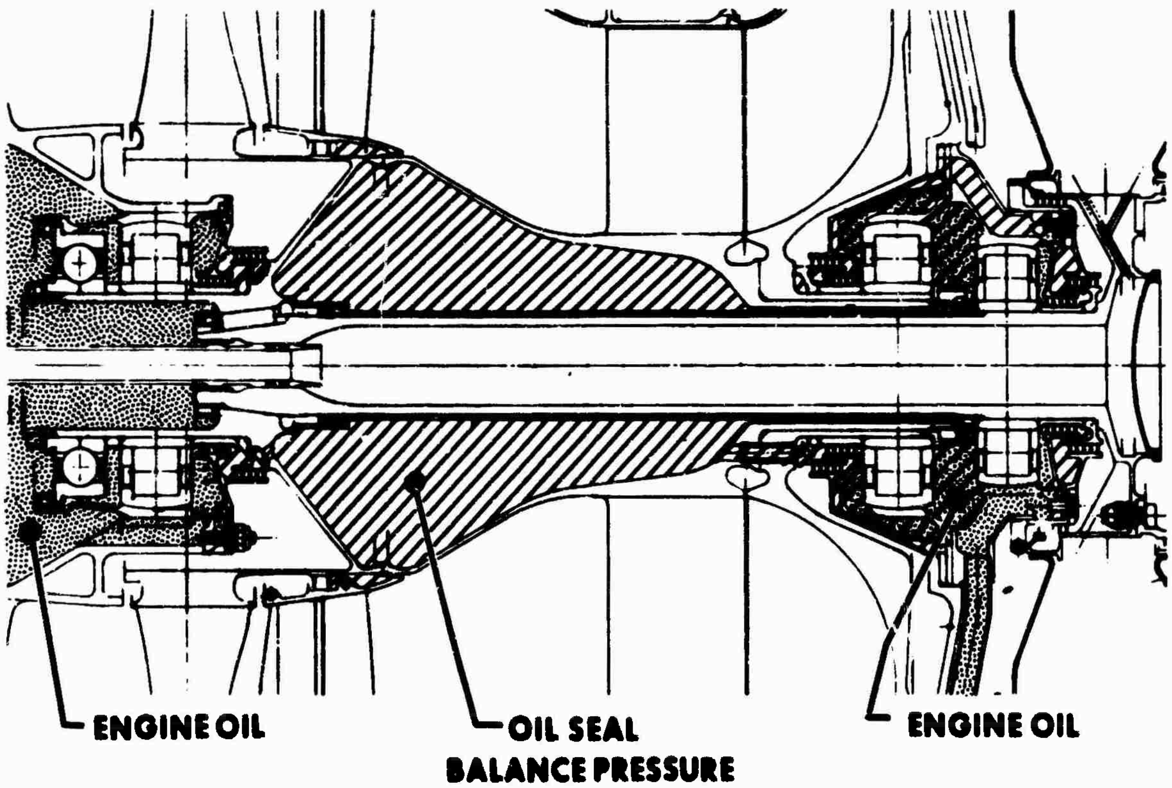


Figure 33. Continental 357-1 Engine Oil Seal Balance Pressure.

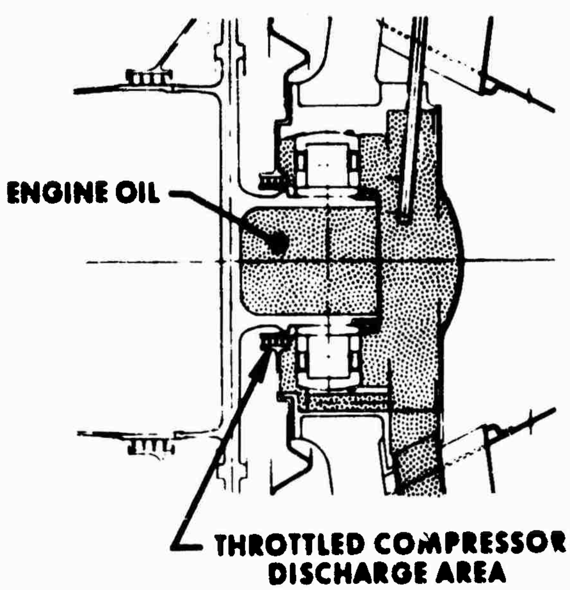


Figure 34. Continental Model 357-1 Rear Bearing Oil Seal.

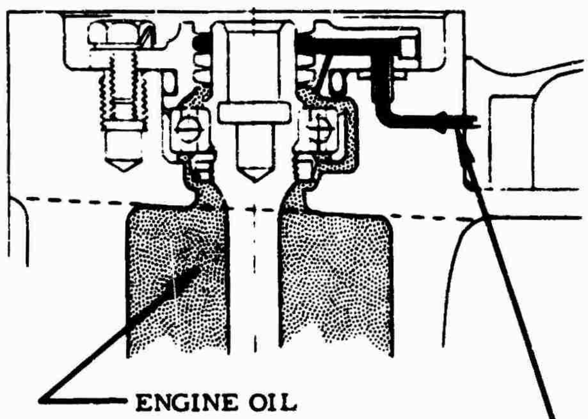


Figure 35. Continental Model 357-1 Fuel Pump and Fuel Control Labyrinth Seal.

Throttled compressor discharge air is provided between the seal lands for the fuel pump and the fuel control labyrinth seal, Figure 35. This air supply provides a continuous airflow into the front bearing compartment.

The air flows through the labyrinth seals into the engine oil compartments, flows through the oil scavenge line to the oil tank, and is exhausted through the oil tank vent to the atmosphere. The designed airflow from the air side into the oil compartment produces low oil seal leakage.

ENGINE WEIGHT AND CENTER OF GRAVITY

The breakdown of engine weight and center of gravity for each major section is presented in Table 7.

TABLE 7

WEIGHT AND CENTER OF GRAVITY SUMMARY

Part No.	Part Name	Weight (lb.)	Horiz. Arm (in.)	Horiz. Mom't. (in-lb.)	Vert. Arm (in.)	Vert. Mom't. (in-lb.)	Aft Arm (in.)	Aft Mom't. (in-lb.)
TURBINE SHAFT ASSEMBLY								
709367	Turbine Rotor and Intermediate Shaft	17.9						
709853	Shaft, Front-Turbine	5.9						
	Misc.	0.3						
	Total	24.1	0	0	0	0	28.10	677.21
COMBUSTOR ASSEMBLY								
709471	Shell Assembly - Outer	16.5						
709365	Nozzle, Inlet - Turbine	35.7						
708364	Housing - Combustor	26.3						
	Misc.	2.5						
	Total	81.0	0	0	0	0	26.80	2,170.80
EXHAUST DUCT AND REAR BEARING								
709374	Duct - Exhaust	26.0						
709973	Adapter, Scavenge Oil	0.3						
306221	Bearing Roller - Turbine	4.9						
	Misc.	3.0						
	Total	34.2	0	0	0	0	33.70	1,152.54

TABLE 7 (Cont'd)
WEIGHT AND CENTER OF GRAVITY SUMMARY

Part No.	Part Name	Weight (lb.)	Horiz. Arm (in.)	Horiz. Mom't. (in-lb.)	Vert. Arm (in.)	Vert. Mom't. (in-lb.)	Aft Arm (in.)	Aft Mom't. (in-lb.)
COMPRESSOR ASSEMBLY								
709351	Rotor - Axial	9.0						
709352	Rotor - Inducer	7.9						
709353	Rotor - Compressor	11.0						
709468	Stator - Compressor	3.5						
709355	Housing - Compressor	49.4						
709560	Shroud - Compressor	9.4						
709356	Diffuser - Radial	6.0						
	Misc.	2.2						
	Total	98.4	0	0	0	0	14.70	1,446.48
TURBINE FRONT AND COMPRESSOR REAR BEARING SUPPORT								
709764	Diffuser - Axial	23.0						
709377	Support Assembly - Compressor	3.7						
306226	Bearing Roller - Turbine	3.6						
709361	Shroud - Turbine Bearing	1.4						
306221	Bearing Roller - Compressor	5.0						
	Misc.	3.1						
	Total	39.8	0	0	0	0	19.00	756.20

TABLE 7 (Cont'd)
WEIGHT AND CENTER OF GRAVITY SUMMARY

Part No.	Part Name	Weight (lb.)	Horiz. Arm (in.)	Horiz. Mom't. (in-lb.)	Vert. Arm (in.)	Vert. Mom't. (in-lb.)	Aft Arm (in.)	Aft Mom't. (in-lb.)
AIR INLET ASSEMBLY								
709463	Housing Assembly 0 - Air Inlet	13.0						
709464	Retainer - Compressor Bearing	1.6						
306221	Bearing Roller - Compressor	5.0						
306264	Bearing Ball - Compressor	3.7						
	Misc.	4.5						
	Total	27.8	0	0	0	0	3.90	108.42
ACCESSORY DRIVE AND CONTROLS								
	Fuel Control	20.0						
	Oil Cooler	5.0						
	Oil Pump	5.0						
	Fuel Pump	4.0						
	Line - Fitting and Ignition	8.0						
	Accessory Drive	7.8						
	Misc.	4.5						
	Total	54.3	2.80	-152.04	0.40	+21.72	3.80	206.34
COMPLETE ENGINE TOTALS		359.6		-152.04		+21.72		6,517.99
CENTER OF GRAVITY LOCATIONS			0.42 inch From Engine Centerline Fuel Control Side.		0.06 inch Above Engine Centerline.		18.13 inches From Front Face of Air Intake Duct.	

FUEL SYSTEM

This section of the report describes only the fuel system contained in the engine. The fuel pump and fuel control are discussed in Heavy Lift Tip Turbojet Rotor System, Volume XII, CAE Report No. 943.

The starting fuel is controlled by a solenoid valve. The starting fuel, Figure 36, is injected through two starting fuel nozzles,

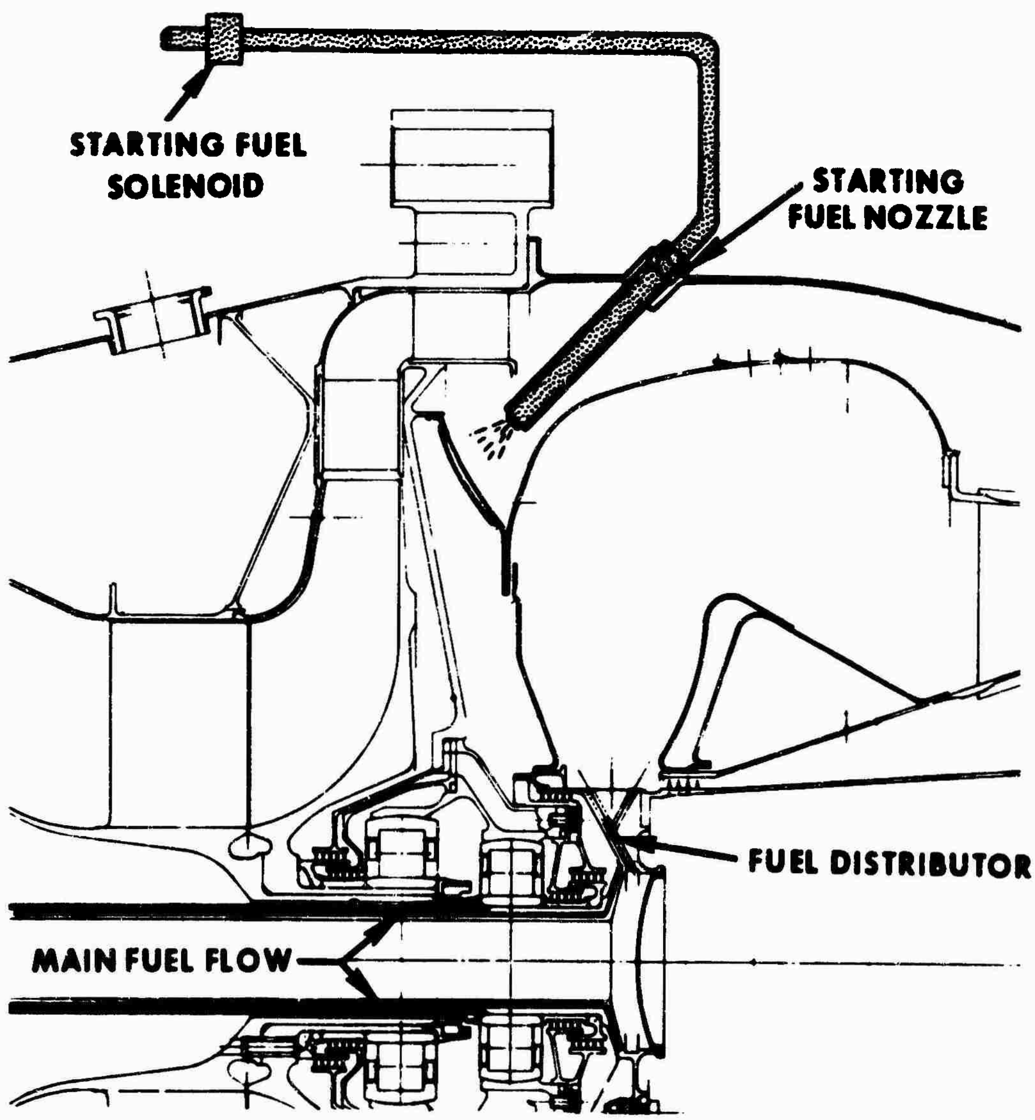


Figure 36. Continental Model 357-1 Engine Starting Fuel System.

placed alongside two igniters. The flame from the starting fuel nozzle spray passes through louvers in the primary swirl vane into the combustor and ignites the main fuel flow issuing from the fuel distributor. After the main fuel flow is ignited, the starting fuel flow is turned off.

The g field is felt as fuel pressure prior to the fuel leaving the spray nozzle. After leaving the spray nozzle, the fuel is not subjected to the g field and should produce a spray pattern similar to a conventional engine.

The metered engine main fuel, Figure 37, enters the engine at the fuel control mounting pad (A) and passes through drilled passageways to the rotating fuel tube (B). From the fuel tube the fuel is sprayed through 18 holes in the fuel slinger (C) into the combustion chamber.

A labyrinth fuel tube seal, Figure 38, is used. This type of fuel tube seal has been employed successively on a previous Continental engine. Fuel enters a low pressure fuel nozzle (1) and sprays into the rotating fuel tube (2). High pressure compressor discharge air is provided at the labyrinth seal annulus (3), from where it leaks past two lands of the seal and enters the rotating fuel tube at (4). Any fuel leakage past the labyrinth seal enters the annulus (5) and is drained overboard. The oil from the main thrust bearing is sealed from the fuel system by a three-land seal (6).

The fuel inside the small rotating fuel tube is subjected to a force of 3400g at full engine r.p.m. The fuel at the fuel slinger is subject to 38,500g. These high g fields should sufficiently overpower the helicopter g field to ensure that fuel tube and fuel slinger operation will be essentially the same as it is in existing Continental engines, and it can be predicted on this basis.

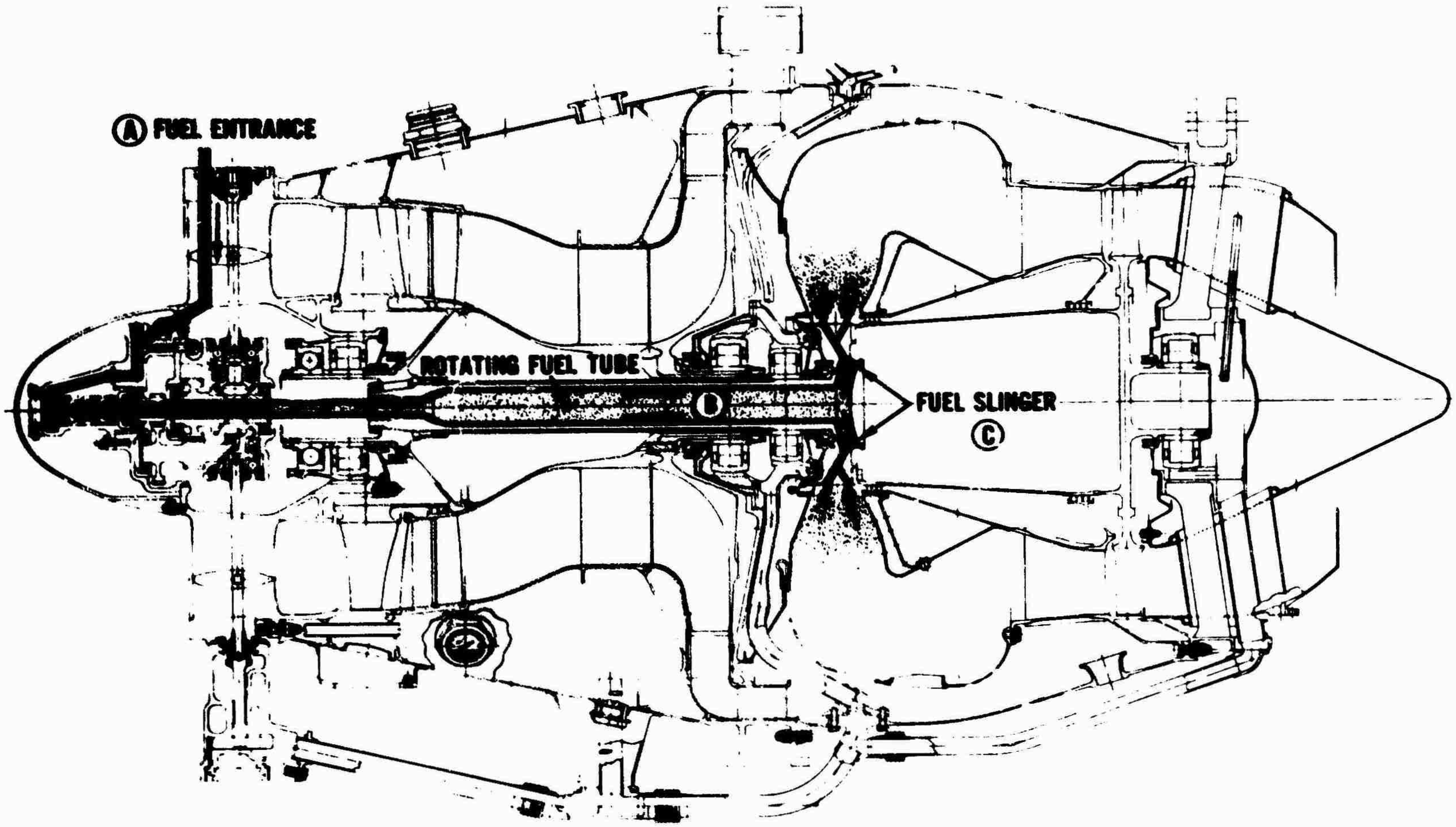


Figure 37. Continental Model 357-1 Engine Main Fuel System.

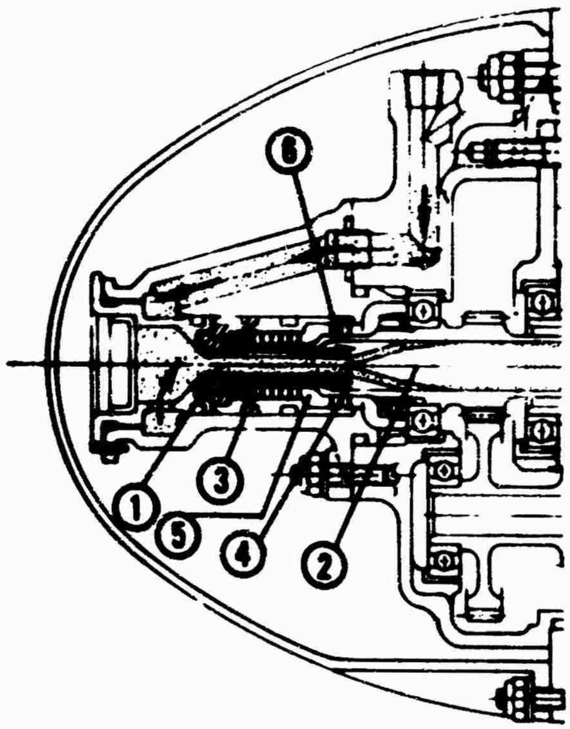


Figure 38. Continental Model 357-1 Engine Fuel Tube Seal.

LUBRICATION SYSTEM

GENERAL DESCRIPTION

Figure 39 is a schematic of the lubrication system for the helicopter rotor tip-mounted gas turbine engine. The system consists of an engine driven pressure pump assembly, an integral oil tank, an engine-mounted heat exchanger (optional), a pressure oil distribution manifold, and a scavenge oil manifold. The system is a "hot tank" system, where the hot scavenge oil is returned directly to the oil tank. The oil is gravity fed from the oil tank to the pressure pump assembly where it is pressurized, filtered, and regulated and then passed through an air-oil heat exchanger, and from there distributed to the various lubrication points on the engine. The oil flow at each lubrication point is controlled by an orifice sized in relation to the available pressure. The estimated heat rejection oil flow rate, and temperature rise at each bearing location is shown in Figure 40: The oil scavenge at each bearing location will be accomplished by the head difference between the bearing sump oil level and

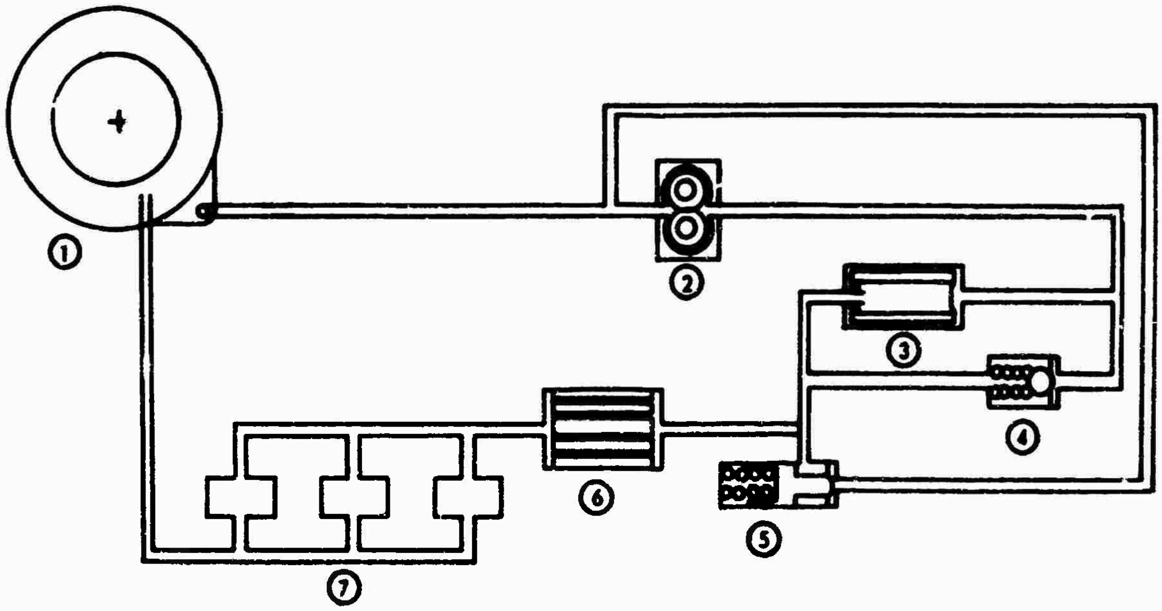


Figure 39. Schematic Diagram of Continental Model 357-1 Lubrication System.

the tank oil level, plus a pressure boost from the labyrinth seal air leakage. Figure 41 shows the engine maximum heat rejection to the oil throughout the engine speed range.

SYSTEM COMPONENTS

Oil Tank

The lubrication oil is contained within a tank integral with the titanium compressor housing assembly. By utilizing the compressor housing weldment, the tank is an integral part of the engine structure. An inner wall is added near the rear of the tank to prevent the oil from coming directly in contact with the hot radial compressor housing. The use of the engine structure provides a large volume oil tank with low weight. The oil tank volume of 24 quarts completely surrounds the engine compressor. The design oil capacity is 6.0 quarts.

FRONT SECTION BEARINGS AND ACCESSORY DRIVE

BEARING DESCRIPTION	ORIFICE NUMBER	ORIFICE LOCATION	FLOW AT 235g (g.p.m.)	HEAT REJECTION	OIL TEMP. RISE	ORIFICE OPERATING PRESSURE (1g)	ORIFICE OPERATING PRESSURE (235g)	ORIFICE SIZE (in. dia.)
FRONT	1	REAR SIDE	.35			40	76.8	.045
COMPRESSOR BEARING	2	FRONT SIDE	.35			40	76.8	.045
THRUST BEARING	3	REAR SIDE	.35			40	76.8	.045
ACCESSORY DRIVE BEARINGS	4	BEVEL GEAR SPRAY	.24			40	80.5	.037
ACCESSORY DRIVE BEARINGS	5	SPUR GEAR SPRAY	.24			40	79.5	.037
TOTALS			1.53	365	64			
FUEL PUMP DRIVE PAD BEARING	6	OUTBOARD SIDE OF BEARING	.17	15		40	91.2	3-.040 IN SERIES
FUEL CONTROL DRIVE PAD BEARING	7	OUTBOARD SIDE OF BEARING	.11	15		40	26.8	2-.040 IN SERIES
CENTER SECTION BEARINGS								
REAR	8	FRONT SIDE	.316			40	80	.043
COMPRESSOR BEARING	9	REAR SIDE	.633			40	80	.062
FRONT	10	FRONT SIDE	.633			40	80	.062
TURBINE BEARING	11	REAR SIDE	.316			40	80	.043
TOTALS			1.898	458	64			
REAR SECTION BEARINGS								
REAR	12	FRONT SIDE	.4			40	80	.048
TURBINE BEARING	13	REAR SIDE	.4			40	80	.048
TURBINE SHAFT COOLING	14	TURBINE SHAFT	.49			40	80	.053
TOTAL			1.29	346	69			
ENGINE TOTAL			4.99	1199	64			

Figure 40. Estimated Heat Rejection, Oil Flow Rate, and Temperature Rise - Continental Model 357-1 Engine.

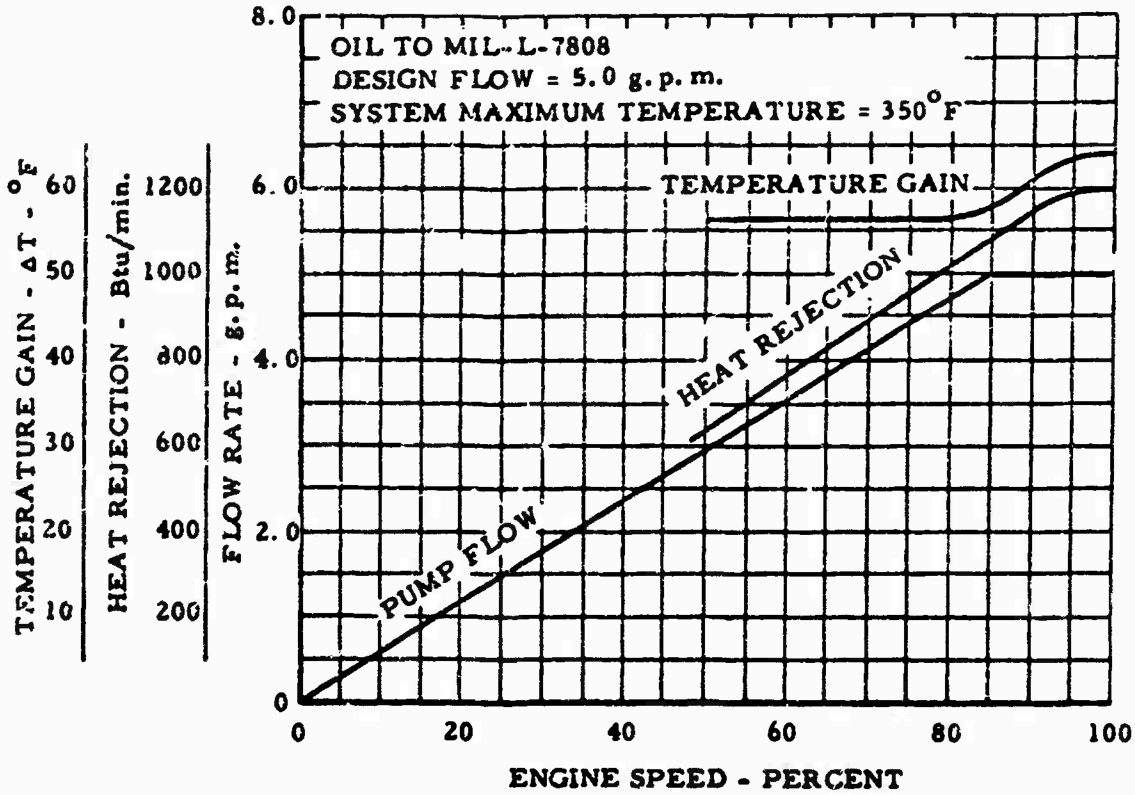


Figure 41. Continental Model 357-1 Lubrication System Characteristics Versus Engine Speed.

Figure 42 shows the oil tank capacity at various oil levels measured from the engine centerline. With an oil volume of six quarts and a system flow rate of 5.0 g.p.m., the oil dwell time is 18 seconds. The 18-second dwell time, in conjunction with the 235g field and large tank volume, provides excellent deaeration characteristics. The oil tank is located around the periphery of the compressor. The connection for the filler, vent, sight gage, drain, and tank outlet are provided by welded bosses at the required location.

The tank outlet is located at 45 degrees below the horizontal on the outboard side of the engine near the major diameter of the oil tank. The tank outlet fitting is shaped so as to be near the lowest point of the tank in both g fields.

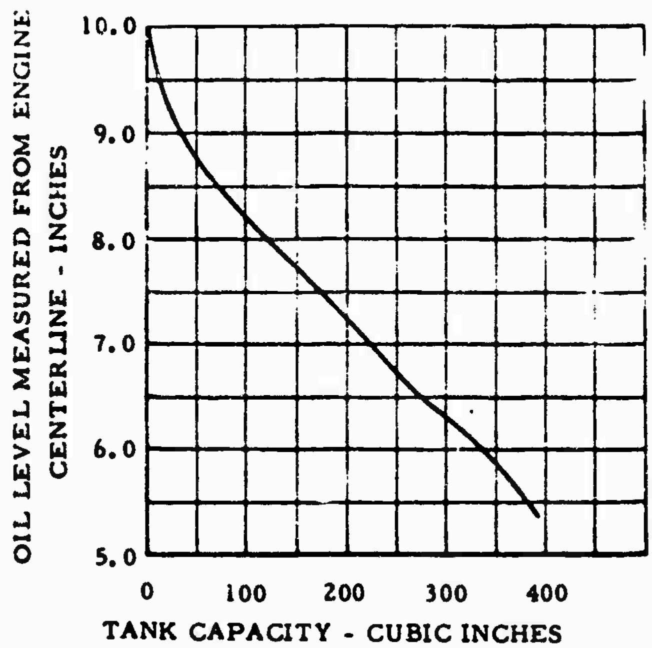


Figure 42. Continental Model 357-1 Lubrication Oil Tank Capacity.

The sight gage is located on the inboard side of the oil tank for visual oil quantity check. The sight gage is mounted in the tank skin so that the oil is not trapped in the gage. At the 6-inch oil level, the sight gage is capable of indicating $\pm$ 0.625 inch of oil, which corresponds to plus 1.0 quart or minus 1.2 quarts.

The oil tank fill port is located 30 degrees from the horizontal on the inboard side. The fill port has a 1-3/8-inch diameter opening, which is ample for hand filling the tank. The fill port cap is sealed with an "O" ring and locked in place by a commercial, hand-actuated locking device.

The tank vent is an AND 10050-16 boss located 30 degrees from the horizontal on the inboard side of the engine. The tank vent is 15 inches from the oil level in the 235g field and 11.2 inches in the one g field. The high location for the tank vent should provide a very low oil entrainment at the vent. A static air-oil separator and vent line can be provided to suit the installation.

The tank drain plug is located directly at the bottom of the oil tank. The 3/4-inch diameter drain opening will provide adequate flow area to permit the oil to drain out in a short period.

Pressure Pump Assembly

The pressure pump assembly contains a gerotor pump element, filter, filter bypass valve, and a pressure regulating valve. The pumping element is sized to displace 0.26 cubic inch per revolution and at 6565 r.p.m. will produce 7.4 g.p.m. at 100 percent volumetric efficiency. In order to deliver 5.0 g.p.m. to the engine, a minimum volumetric efficiency of 67.5 percent is needed. Past experience indicates that better than 90 percent volumetric efficiency may be obtained with these pumps at standard sea level conditions. The volumetric displacement is adequate to allow for pump flow loss due to pressure, temperature, speed, and wear. Since the pump housing has a high coefficient of thermal expansion, a ring that has the same coefficient of expansion as the pump elements has been installed in the housing to control the clearance variation with operating temperature. The pump inlet "kidney" is located so that a smooth flow results from the oil tank outlet line.

The filter is a standard pleated construction with the filter media backed up with a coarse-weave screen, a support spring, and a perforated tube to provide for rigidity in the longitudinal direction and high level crush strength. The filter rating is 64 microns at 100 percent effective. The effective area of the filter is 31 square inches, resulting in a low pressure drop of 0.6 p.s.i. at 100°F with sufficient area margin for dirt build-up.

The valves are oriented so that the valving action is parallel to the g field (ref. Figure 39). The force on the valves due to the weight of the valve and spring acted upon by the high g load will increase the valve operating pressure, which is initially set by the valve spring. This characteristic is used to advantage in the pressure regulating valve to provide the increased pump discharge pressure required for operation in the high g field. The weight of the valve is controlled such that, in conjunction with the increasing g load, the desired system pressure is maintained. The 1g pump output pressure is set at 40 p.s.i. and at the 235g condition the pump pressure increases to 176 p.s.i.

The filter bypass valve is a hollow stainless steel ball for low weight to minimize the bypass pressure increase in the high g field. The 1g bypass pressure differential of 10 p.s.i. increases to 16 p.s.i. in the 235g field.

Pressure Distribution System

The lubrication oil for the accessory drive and front engine bearings will be carried down an inlet strut and distributed by five orifices. Lubrication oil for the fuel pump drive pad bearing and the fuel control drive pad bearing is carried around the outside of the front support ring by cored holes to the respective drive pads. The center bearing and the rear bearing lubrication oil is carried to the respective locations by external lube lines. Figure 43 shows the pressure drop in the oil distribution lines at 75^{o} and $250^{o}F$.

Scavenge System

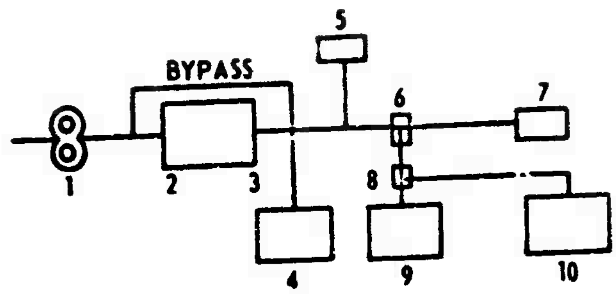
At each lubrication point, a sump area is provided to collect the oil within the bearing cavity. The engine rotor bearings have sufficient scavenge area around the bearing to ensure that oil level build-up next to the labyrinth seal is not possible. The sump drain lines are located on the outboard side below the horizontal to provide for oil scavenge in the right angle g fields. Figure 44 shows scavenge line pressure drop at various flow rates and temperatures.

During start up, and in the absence of the high g field, the oil flow rate at engine idle is reduced to 70 percent normal due to a reduced orifice operating pressure. When the helicopter rotor r.p.m. is increased, the oil flow increases and the scavenge flows are given a boost by the increasing g field. At 235g an oil level difference of one inch is equivalent to 7.9 p.s.i.

HEAT EXCHANGER

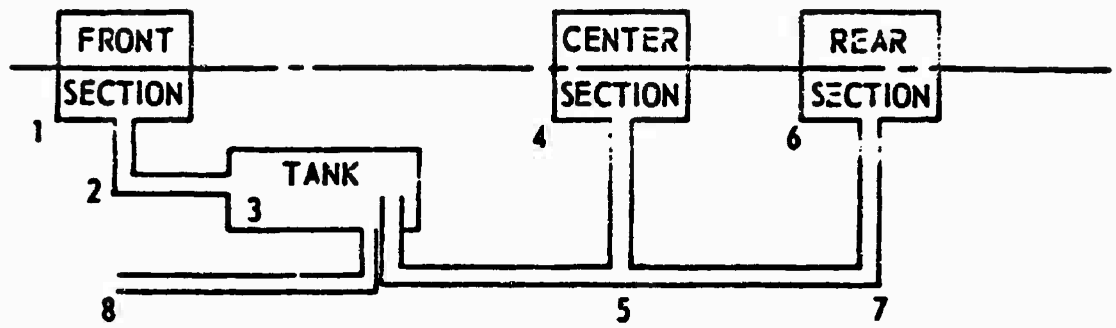
Provisions have been made for an engine mounted air-oil heat exchanger to accommodate the engine heat rejection of 1200 B.t.u. per minute. A fuel-to-oil heat exchanger must be ruled out on the basis that the engine heat rejection would boil the fuel in the low pressure fuel system employed on this engine.

The cooler mounting pad is provided on the outboard side of the engine front frame and uses six 1/4-inch studs to support the cooler. The inlet and exit ducting would be provided by the airframe manufacturer.



LOCATION	LINE SIZE DIAMETER (INCH)	LINE LENGTH (INCHES)	FLOW (g.p.m.)	PRESSURE DROP p.s.i. AT 75°F	PRESSURE DROP p.s.i. AT 250°F
1-2 PUMP TO HEAT EXCHANGER	.375	4	5	.57	.51
2-3 HEAT EXCH.	-	-	5	-	5
2-3 HEAT EXCH. BYPASS	-	-	5	2	-
3-4 HEAT EXCH. TO FRONT BEARING	.203	5	1.53	1.49	1.33
3-5 HEAT EXCH. TO FUEL PUMP PAD BEARING	.375	10.5	3.47	.86	.72
5-6 FUEL PUMP LOCATION TO FITTING	.375	4	3.30	.28	.25
6-7 FITTING TO FUEL CONTROL PAD BEARING	.188	8.8	.11	.33	.042
6-8 FITTING TO CENTER FITTING	.320	21	3.19	2.61	2.45
8-9 CENTER FITTING TO CENTER BEARINGS	.194	9.3	1.90	5.3	4.85
8-10 CENTER FITTING TO REAR BEARING	.194	19.3	1.29	7.5	4.9

Figure 43. Estimated Pressure Drop in Oil Distribution Lines - Continental Model 357-1 Engine.



LOCATION	EQUIVALENT LINE SIZE DIAMETER (INCH)	LENGTH (INCH)	FLOW (g.p.m.)	ΔP AT 75° (p.s.i.)	ΔP AT 250° (p.s.i.)
1-2 STRUT	.550	6	1.81	.048	.006
2-3 TWO TRANSFER TUBES	.375	5.3	1.81	.098	.013
4-5 CENTER SCAVENGE TUBE	.555	10.8	1.90	.087	.011
6-7 REAR SCAVENGE TUBE	.555	7	1.29	.039	.005
7-5 REAR FITTING CENTER FITTING	.555	14	1.29	.078	.010
5-3 CENTER FITTING TO TANK	.680	13.5	3.19	.078	.010
3-8 PUMP INLET LINE	.680	10	5	.097	.012

Figure 44. Estimated Pressure Drop in Scavenge Lines - Continental Model 357-1 Engine.

The present lube system design dictates that the cooler be mounted outboard of the vertical engine centerline. Coolers may be mounted inboard of this centerline but would require modification to the lube system.

Studies indicate that cooler size and weight can vary considerably as a function of the cooling air pressure drop. An aluminum cooler design with 6 to 8 inches of water air pressure drop would require about 40 square inches of area and weigh about 5 pounds including an automatic oil temperature regulating valve. The required airflow would be about 2.0 pounds per second. The ram drag on such a cooler would be approximately 40 pounds without any recovery, but with recovery this drag could be either totally eliminated or greatly reduced.

STARTING SYSTEM

The CAE Model 357-1 engine is designed for air starting. The axial compressor, Figure 45, is surrounded by an annular manifold (1) with a number of nozzles for air impingement on the blades of the axial compressor. The starting air is supplied to the engine through a flanged opening (2).

The axial compressor air starting characteristics are shown in Figure 46. This curve is based on actual starting tests conducted on a similar Continental engine. The recommended minimum light-off speed is 2000 r.p.m. Although successful starts have been accomplished at lower speeds, the high turbine inlet temperatures experienced would dictate that lower starting r.p.m.'s must be considered only as an emergency measure.

The estimated windmill starting envelope is shown in Figure 47. It is planned to investigate various means of extending this envelope during Phase II of the development. The target for the starting capability is to be able to make windmill starts at forward flight speeds up to 900 feet per second engine forward speed. This figure is greater than the flight speed experienced by the advancing engines during maximum helicopter flight speed at maximum rotor speed.

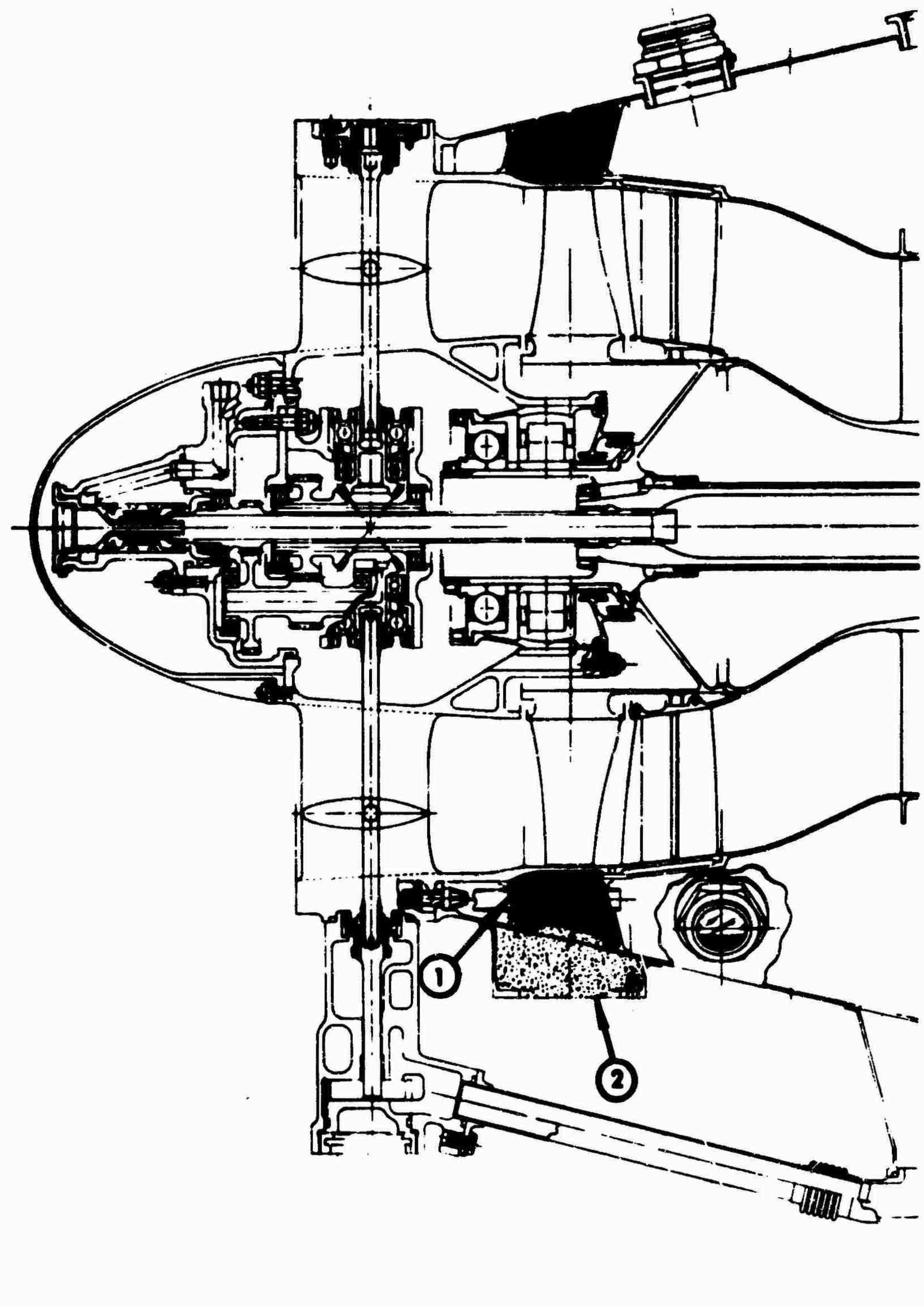


Fig. 45. Continental Model 357-1 Air Starting System.

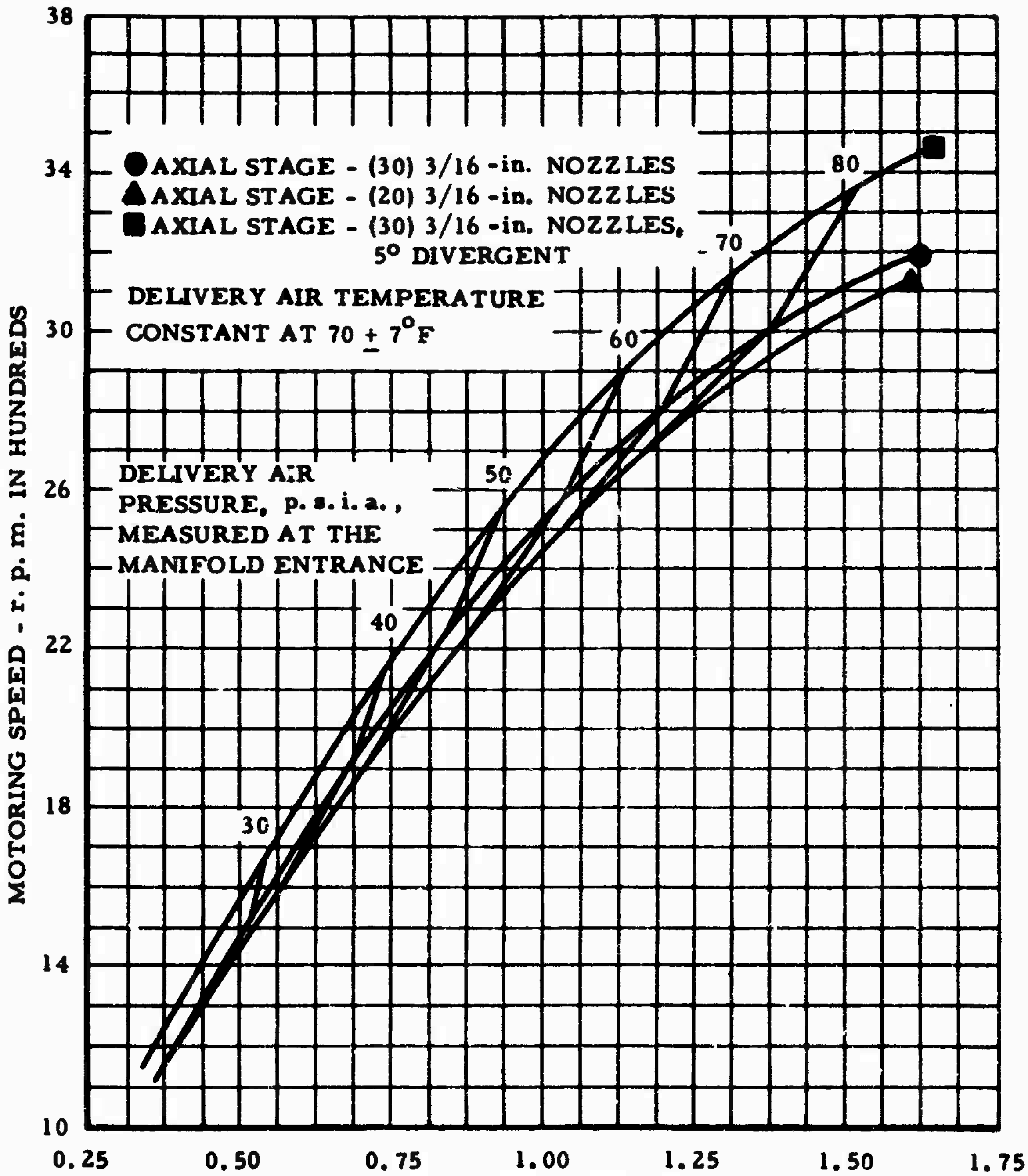


Figure 46. Continental Model 357-1 Axial Compressor Starting Characteristics.

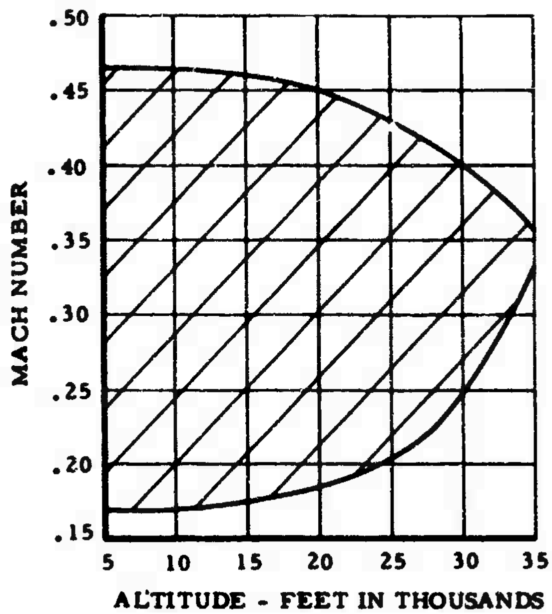


Figure 47. Continental Model 357-1 Estimated Windmill starting Envelope.

ANTI-ICING

Provisions have been made to provide an anti-iced air intake for the engine. The anti-icing system shown in Figure 48 is very similar to the system presently in use by Continental on a similar intake.

Compressor discharge air will be bled from the combustor housing (1) to an annular manifold (2) around the outside of the air intake. From the outside manifold air is piped down the five struts in the air intake to an annular manifold (3) and to the nose cone (4). The placing and number of bleed holes in the engine intake will be determined during subsequent engine testing, and is dependent upon the airflow required to provide sufficient heat to the parts requiring de-icing.

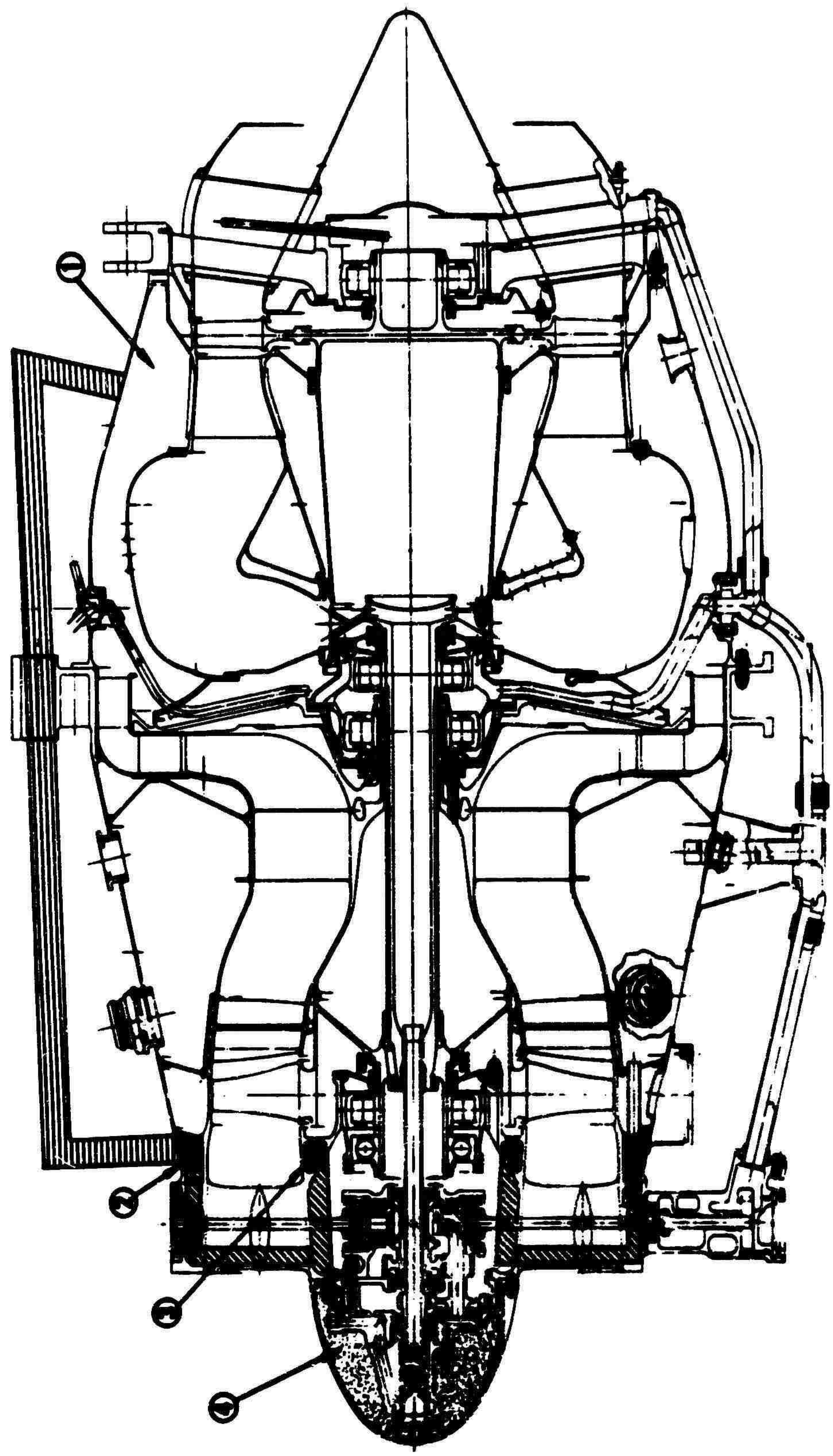


Figure 48. Continental Model 357-1 Engine Inlet Anti-Icing System.

A very simple electrical system, having a minimum number of components, is used on the Model 357-1 engine. A 24-volt DC electrical supply is required to the ignition coil and the starting fuel solenoid, which are used only during engine starting. In addition to these two items, the only other electrical connections on the engine are to the gas temperature-measuring thermocouples, the power supply to the fuel control, and the tachometer instrumentation.

The complete electrical system, Figure 49, consists of an igniter coil, two igniter plugs, and an electrically operated primer fuel solenoid. The igniter leads, coil, and primer solenoid are shielded in accordance with Military specifications to prevent radio interference.

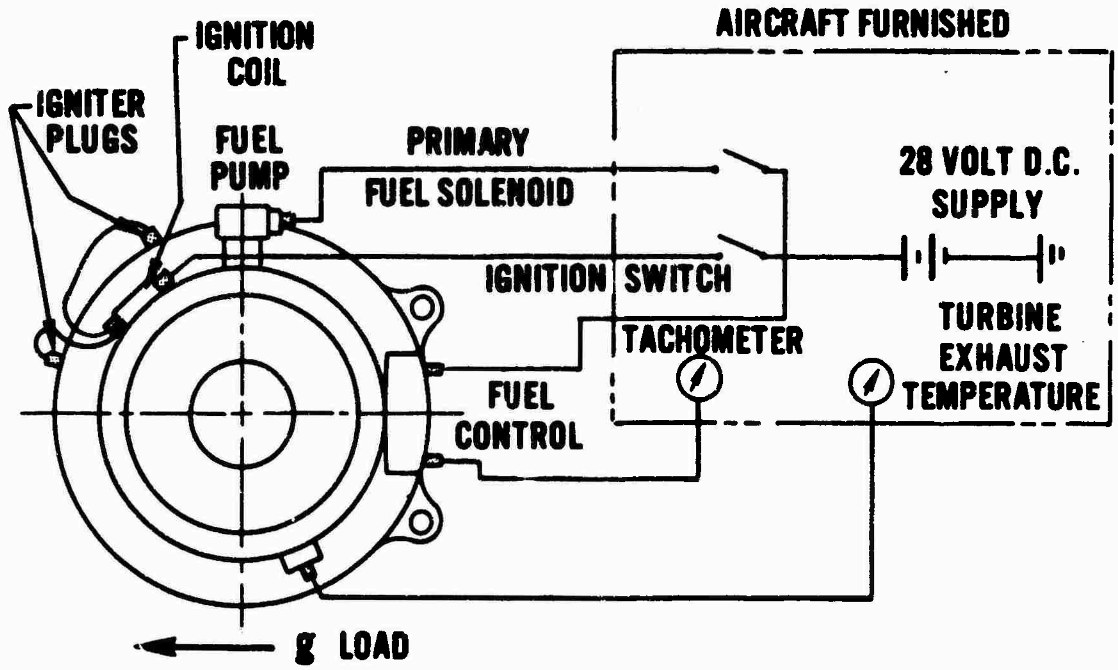


Figure 49. Schematic Diagram of Continental Model 357-1 Electrical System.

Preliminary negotiations with Scintilla Division of Bendix Corporation have indicated the feasibility of using a conventional capacitance discharge spark splitter turbine ignition system in a 259g field.

The positioning of the vibrator mechanism, relative to the direction of the g field, is considered critical and a development and endurance testings program would have to be undertaken to guarantee the reliability of the electrical components in the high g field environment.

The engine is designed with five chromel-alumel thermocouples to measure the temperature of the exhaust gases. These thermocouples are wired together to give a single temperature indication to the pilot.

SECTION FIVE. AERODYNAMIC DESIGN

SUMMARY

The service-proven aerothermodynamic design of the J69-T-29 was retained virtually intact for the Model 357-1 tip turbojet engine. While the mechanical design of the aerodynamic components was affected by the operating environment peculiar to the application, the mechanical design changes were made in such a manner as to not affect the basic aerodynamic performance of the component except as noted below.

Consideration was given to the fact that the J69-T-29 was a short life engine for drone applications and the Model 357-1 was to be a long-life engine for use on man-carrying aircraft. Compliance with this basic change in requirements dictated lower turbine inlet temperatures than those experienced by the J69-T-29. Accordingly, as the mechanical design progressed, some minor aerodynamic changes were accomplished simultaneously in order to increase the mass flow and compressor pressure ratio of the Model 357-1. These changes reduced the predicted turbine inlet temperature to acceptable levels for a long-life engine, and reduced the specific fuel consumption to a value that will permit more economical operation of the helicopter.

During this phase, tests were run on the J69-T-29 engine to establish the effects of inlet distortion on engine performance. The results of this testing will be combined with the results of the Phase II wind tunnel tests of the Hiller-designed engine nacelle inlet to predict engine performance in the vehicle under flight conditions.

A preliminary engine specification was prepared which reflects the performance of the fully qualified engine. The prototypes used during the development conducted under Phase II of the program are expected to exhibit performance that agrees generally with this specification except in specific fuel consumption. At this stage of the development, we would expect the specific fuel consumption of the prototype to be between 1.03 - 1.06.

The primary differences between the prototype engines and the qualification engines are expected to be in the centrifugal compressor and the combustor. The introduction of a compressor of the type shown in Figure 52 should raise the overall compressor efficiency by approximately 5 percent and the introduction of a combustor of 1 percent less pressure loss along with the reduction of cycle temperature at constant thrust that these component improvements would permit, should enable the qualification engine to meet its guarantees.

DISCUSSION

PURPOSE

The primary purpose of the aerothermodynamic design work conducted under Phase I of the tip turbojet development was to evaluate the effects of the operating environment on engine performance, and to recommend the component changes necessary to accommodate required mechanical changes without disturbing the aerodynamic performance of the components.

SCOPE

The aerodynamic effect of Coriolis forces, centrifugal forces, and varying inlet pressure recovery distortions were evaluated in making design recommendations. In order to counter the high centrifugal forces in the structural design, it was desirable to reduce engine length to the greatest extent possible, and a thorough evaluation of the effects of shortened flow paths on engine performance was made.

AERODYNAMIC DESIGN EVALUATION

General

The CAE Model 357-1 engine employs the straight-through aerodynamic flow path characteristic of the J69 engine family. Air enters the engine axially and passes through a single-stage transonic axial compressor, a single-stage centrifugal compressor with a radial and an axial diffuser, an annular combustor, a single-stage turbine, diffusing tailpipe, and plug-type jet nozzle.

The prototype engine cross section is presented in Figure 50. Differences from the established J69-T-29 design are most apparent in the jet nozzle and mechanical structure; the other aerodynamic components are similar to, and based upon, established CAE units.

Compressor Section

The axial compressor stage was scaled from a previously tested component selected to increase overall engine life by improving compressor surge margin and increasing mass flow through the engine. This in turn will permit lower operating temperatures consistent with the requirements for lower specific fuel consumption and increased engine life for man-carrying applications. The scaled compressor was further modified to increase the number of blades to 26 and blade chord length was reduced as necessary to maintain the same aerodynamic performance while providing a mechanical design better suited to the operating environment.

The centrifugal compressor stage utilized is aerodynamically the same as that used in the J69-T-29 engine. This stage combined with the axial stage will produce a specific fuel consumption of about 1.04 in the prototype engines to be utilized at the onset of Phase II of the program. Refinements in the centrifugal compressor and the radial diffuser section will occur during the development testing, and at the completion of development, the compressor will perform essentially in agreement with the data presented in Figure 51. These data were obtained from a compressor whose mechanical design is shown in Figure 52. As this performance is achieved, the fuel consumption of the engine will correspond to that shown in the production engine specification; that is, a specific fuel consumption value of 0.99 at the Military rating.

Turbine Section

The turbine stage is the existing J69-T-29 turbine. The exhaust diffuser, tailcone, and jet nozzle have been redesigned for minimum length and weight.

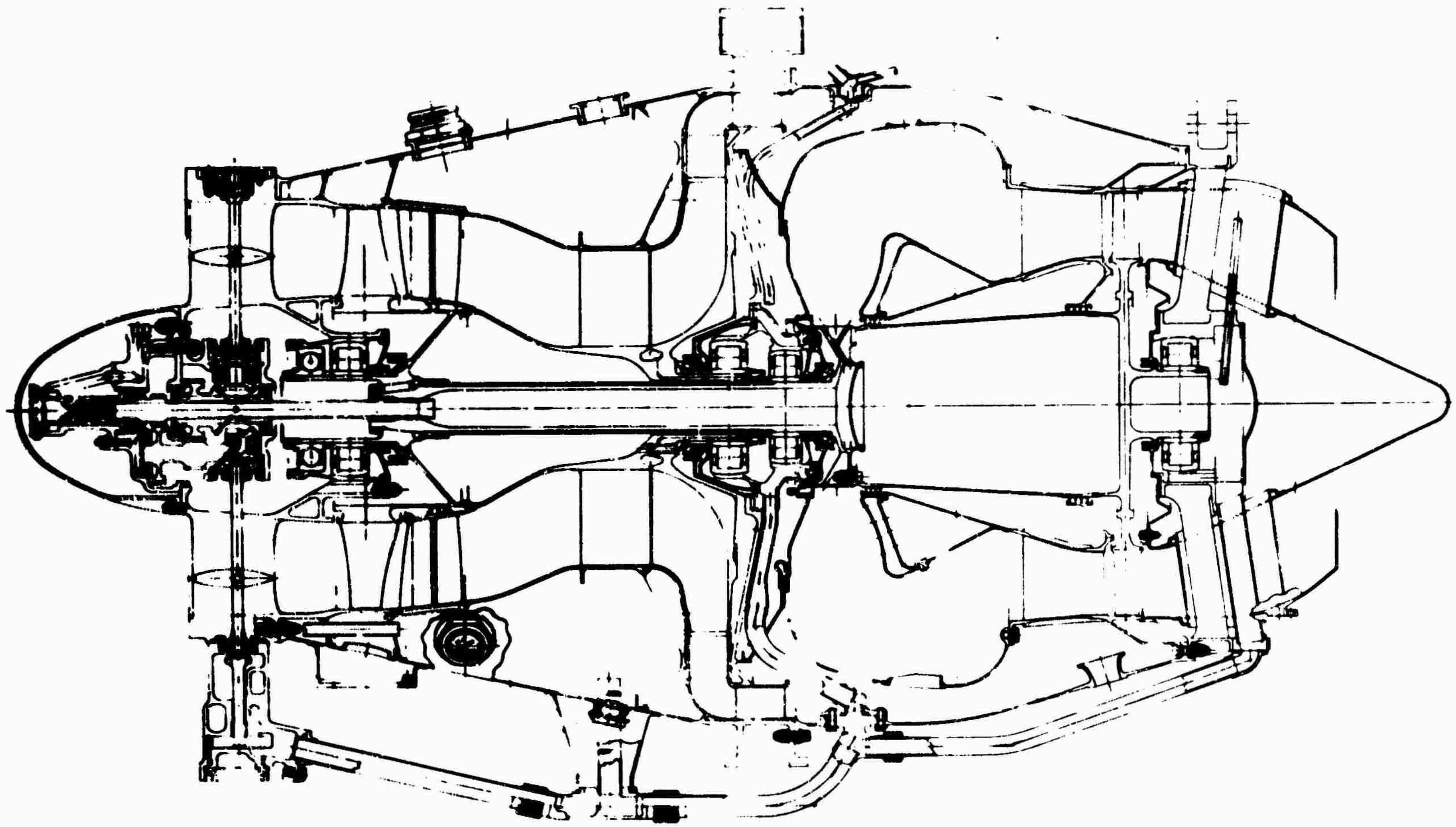


Figure 50. Continental Model 357-1 Engine Cross Section.

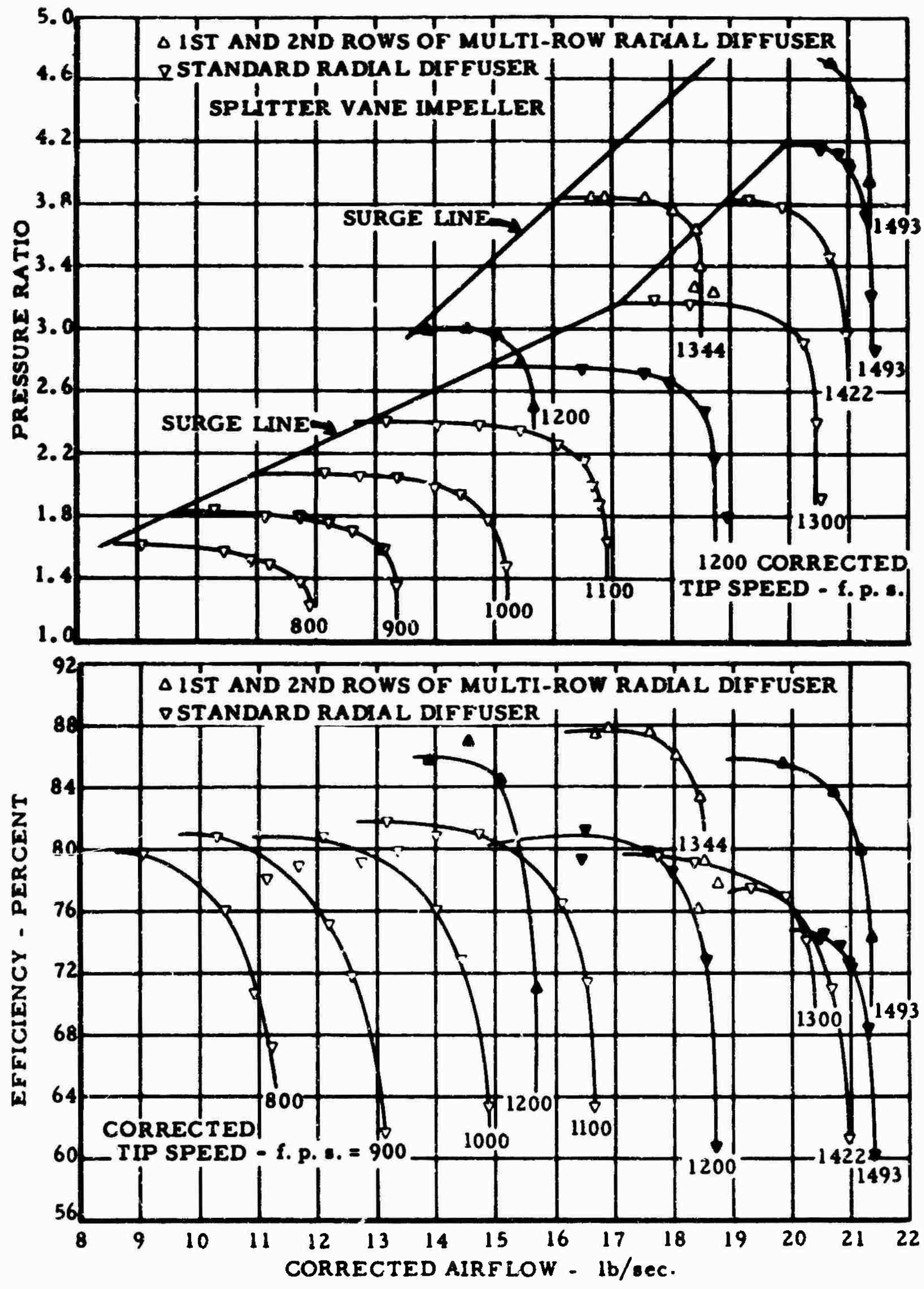


Figure 51. Performance Comparison - Baseline and Two-Stage Diffuser.

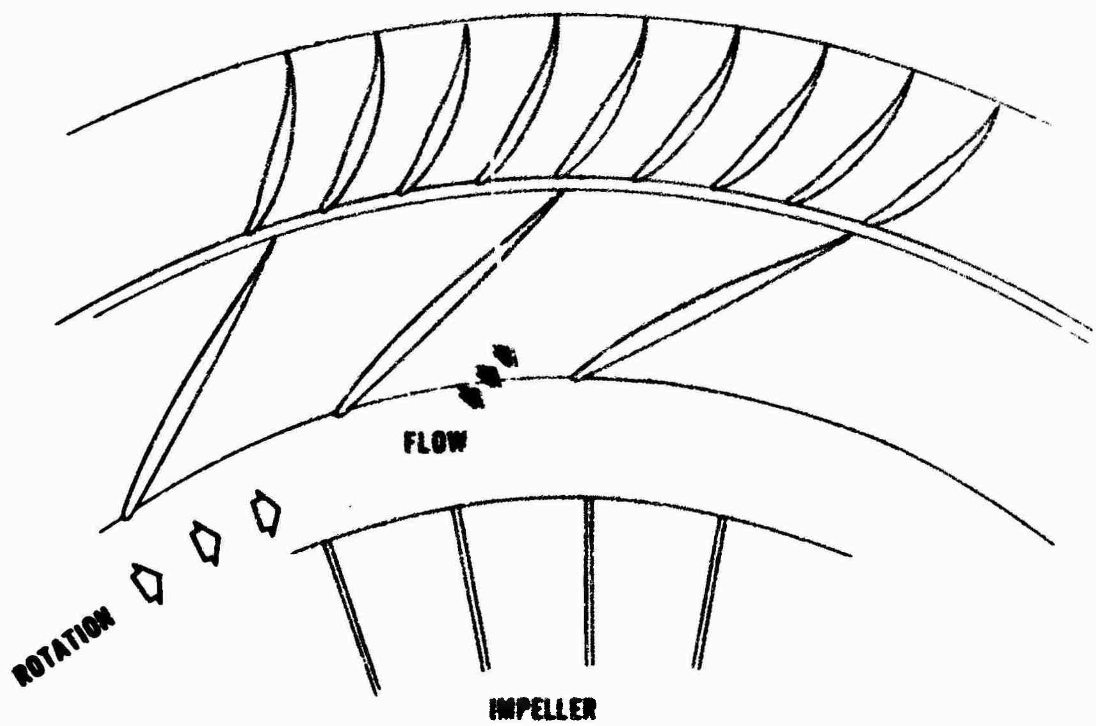


Figure 52. Two-Stage Radial Diffuser Configuration.

Thermodynamic Cycle

The thermodynamic cycle shown in Figure 53 represents the cycle required of the Model 357-1 in order to produce 1700 pounds of thrust at a specific fuel consumption value of 0.99. To attain this cycle, the engine components will require rematching, with the areas and extent of the rematching to be based on the results of the development testing which will be accomplished prior to qualification of the engine for operational use.

Acceleration Characteristics

Figure 54 is the estimated compressor map for the prototype Model 357-1 engine. An estimated operating line and the maximum acceleration limit are also shown. Aerodynamically, this prototype engine is identical to the standard J69-T-29, except for incorporation of an improved flow axial compressor. Therefore, the Model 357-1 engine time constant was computed from prototype and J69-T-29 data adjusted for estimated characteristics. Figure 55 presents the time constant to be used for engine acceleration and response predictions.

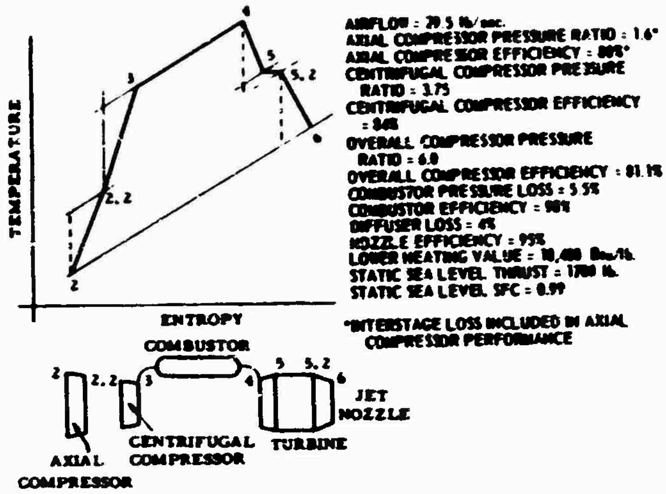


Figure 53. Thermodynamic Cycle and Entropy Diagram.

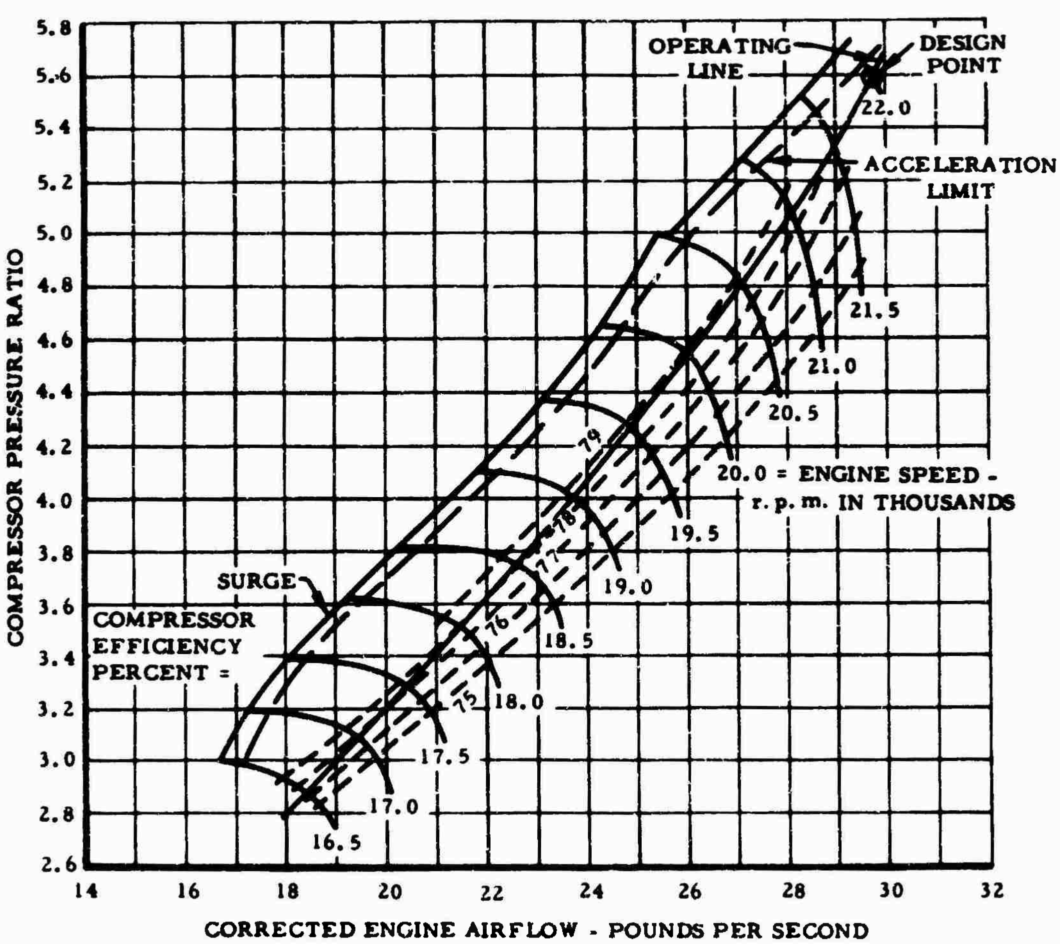


Figure 54. Estimated Compressor Performance - Continental Model 357-1 Engine.

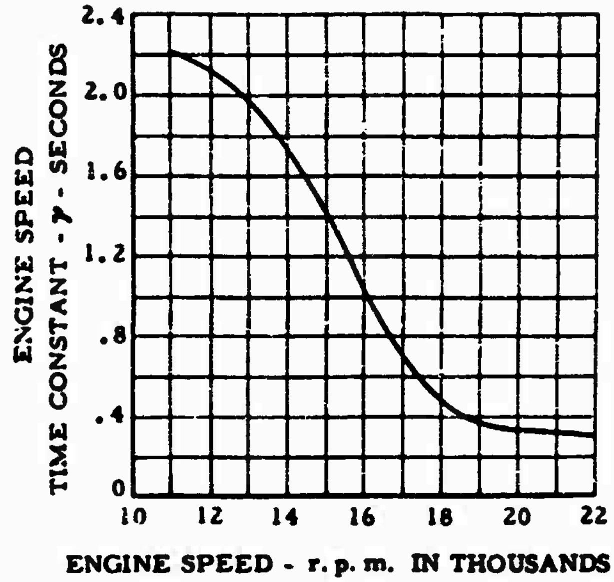


Figure 55. Estimated Engine Time Constant Versus Engine Speed - Continental Model 357-1 Engine.

EFFECT OF INLET DISTORTIONS

The J69-T-29 engine was operated on a sea level static test stand with various measurable inlet distortions. The distortions were accomplished by introducing varying density screens into the engine inlet. Data were gathered to establish the relationship between the engine performance parameters and various degrees of inlet distortion.

The data will be utilized during future test programs. At that time, Hiller Aircraft will run inlet distortion tests on their engine nacelle during wind-tunnel evaluation of the complete nacelle design. Comparison of the two sets of data will permit a precise prediction of installed engine performance under flight conditions.

The following presents a summary of the inlet distortion test.

Object

To evaluate effects of inlet distortion on performance and surge on the CAE Model 357-1 engine.

Results

Performance was established for a standard inlet and two distortion patterns at engine speed of 22,000 r.p.m., as shown in Table 8.

TABLE 8 EFFECT OF INLET DISTORTION ON ENGINE PERFORMANCE			
	Standard Inlet	Screen Pattern "A" (Figure 62) 11.8 Max. -5.8 Avg. Press. Drop	Screen Pattern "B" (Figure 63) 19.5 Max. - 8.8 Avg. Press. Drop
F_n	1690	1540	1390
EGT	1220	1244	1270
SFC	1.055	1.100	1.173
W_f	1770	1700	1630
W_a	29.3	27.4	25.6
P_2/P_1	5.70	5.35	5.05

Surge-free operation was demonstrated with the most severe distortion pattern. The engine was accelerated from idle (11,000 r.p.m.) to maximum speed (22,000 r.p.m.).

Instrumentation

Several different screens were obtained with varying density, Table 9, to develop the desired distortion. An aluminum ring was designed to be mounted between the inlet housing and the air inlet bell for support of the screens and necessary instrumentation.

TABLE 9 TEST SCREENS	
6 x 6 Mesh/Inch,	.035 Wire
50 x 45 Mesh/Inch,	.010 Wire
50 x 50 Mesh/Inch,	.009 Wire
20 x 20 Mesh/Inch,	.015 Wire
8700 Dutch Weave Filter Cloth, 80 x 700 Mesh,	.004 x .003 Wire

The inlet screen configuration is shown in Figure 56. The instrumentation, as observed from the front of the inlet bell, is illustrated in Figure 57. The distortion equipment, as mounted on the engine, is shown in Figures 58 through 61.

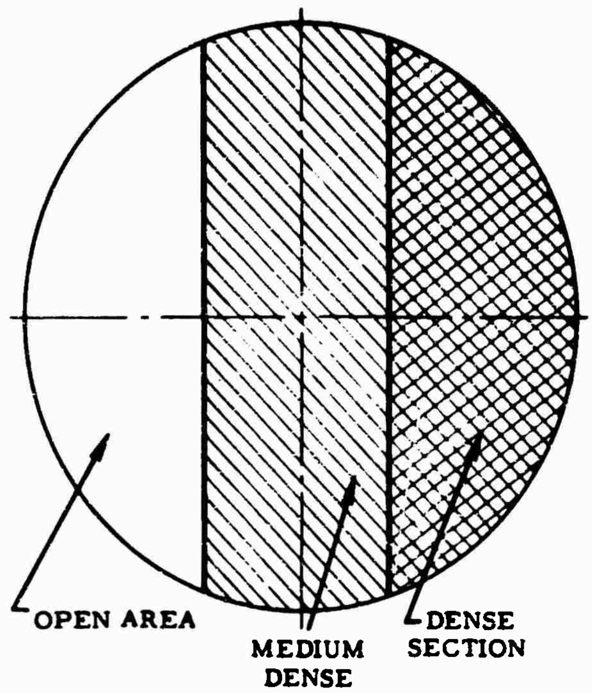


Figure 56. Inlet Screen Configuration

<u>Discussion</u>

This test was proposed to evaluate effects of inlet distortion on performance and surge margin of the Model 357-1 engine. The aerodynamic similarity between the J69-T-29 engine and the Model 357-1 was sufficient to permit use of the J69-T-29 to establish these effects; thus, J69-T-29 engine serial No. 069 was utilized for the test.

Performance was established for a standard inlet condition and two distortion patterns, as follows:

1. Screen Pattern "A," Figure 62, produced a maximum pressure drop of 11.8 percent and an inlet overall pressure drop of about 5.8 percent.

2. Screen Pattern "B," Figure 63, produced a maximum pressure drop of 19.5 percent and an inlet overall pressure drop of 8.8 percent.

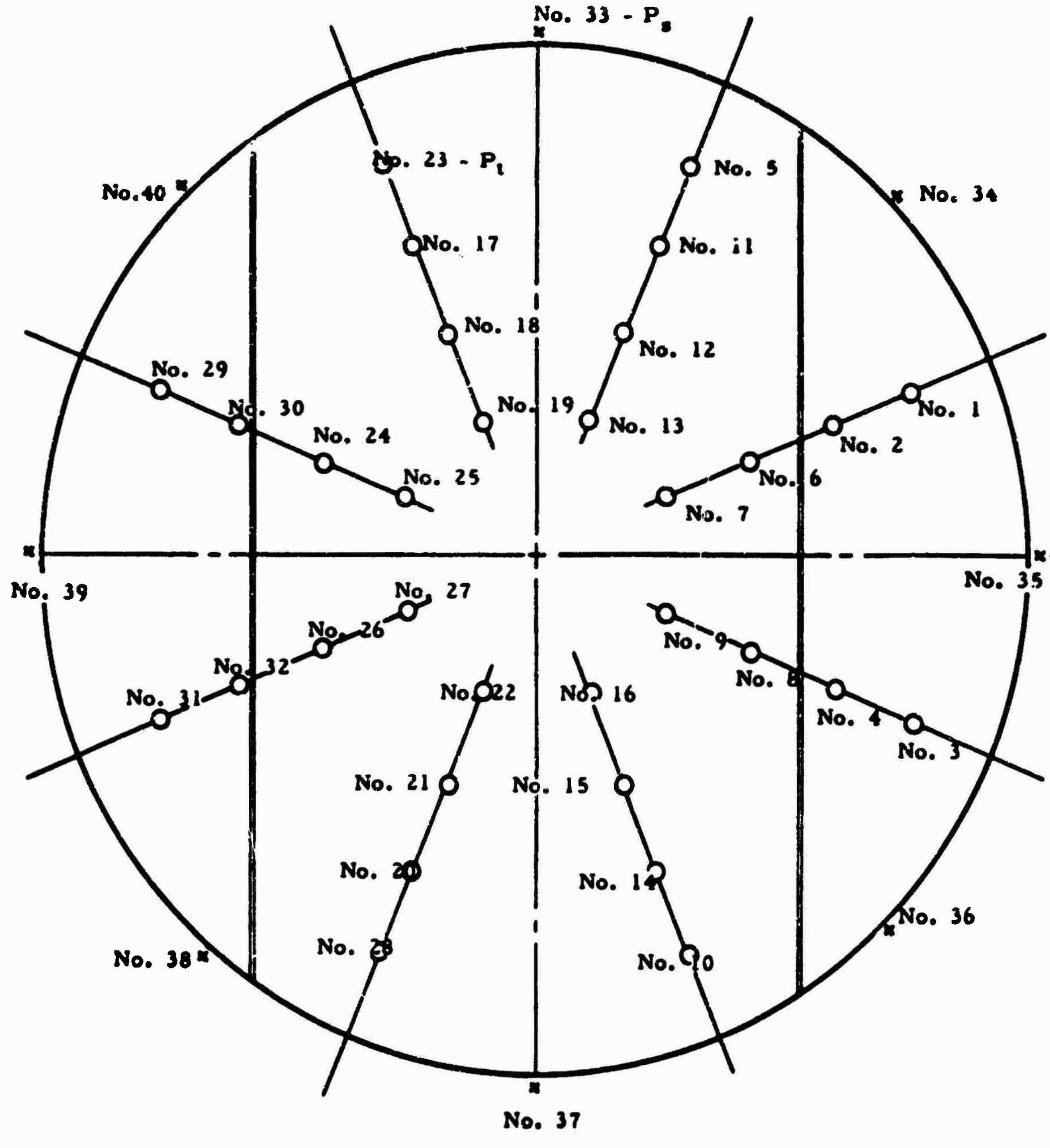


Figure 57. Instrumentation of Air Inlet for Distortion Test.

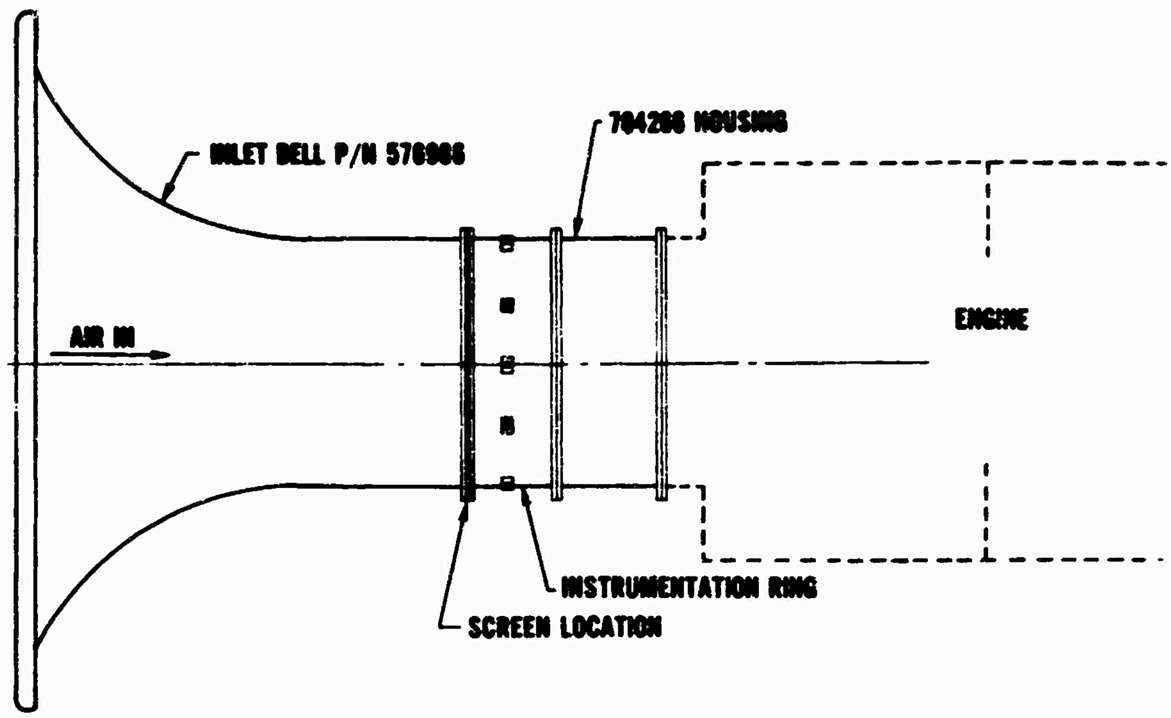


Figure 58. Schematic Diagram of Inlet Distortion Test Instrumentation.

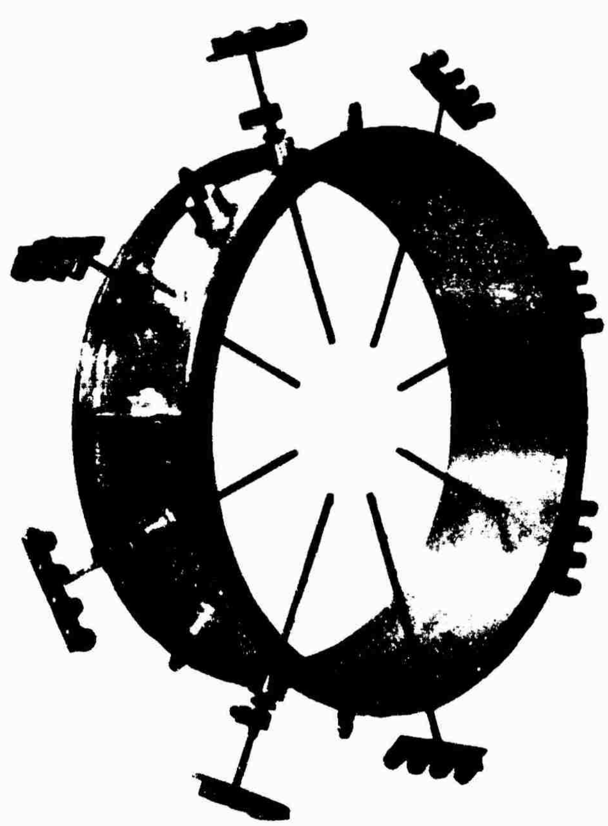


Figure 59. Installation of Static Pressure Probes and Total Pressure Rakes on Adapter Ring - Model 357-1 Inlet Distortion Test.

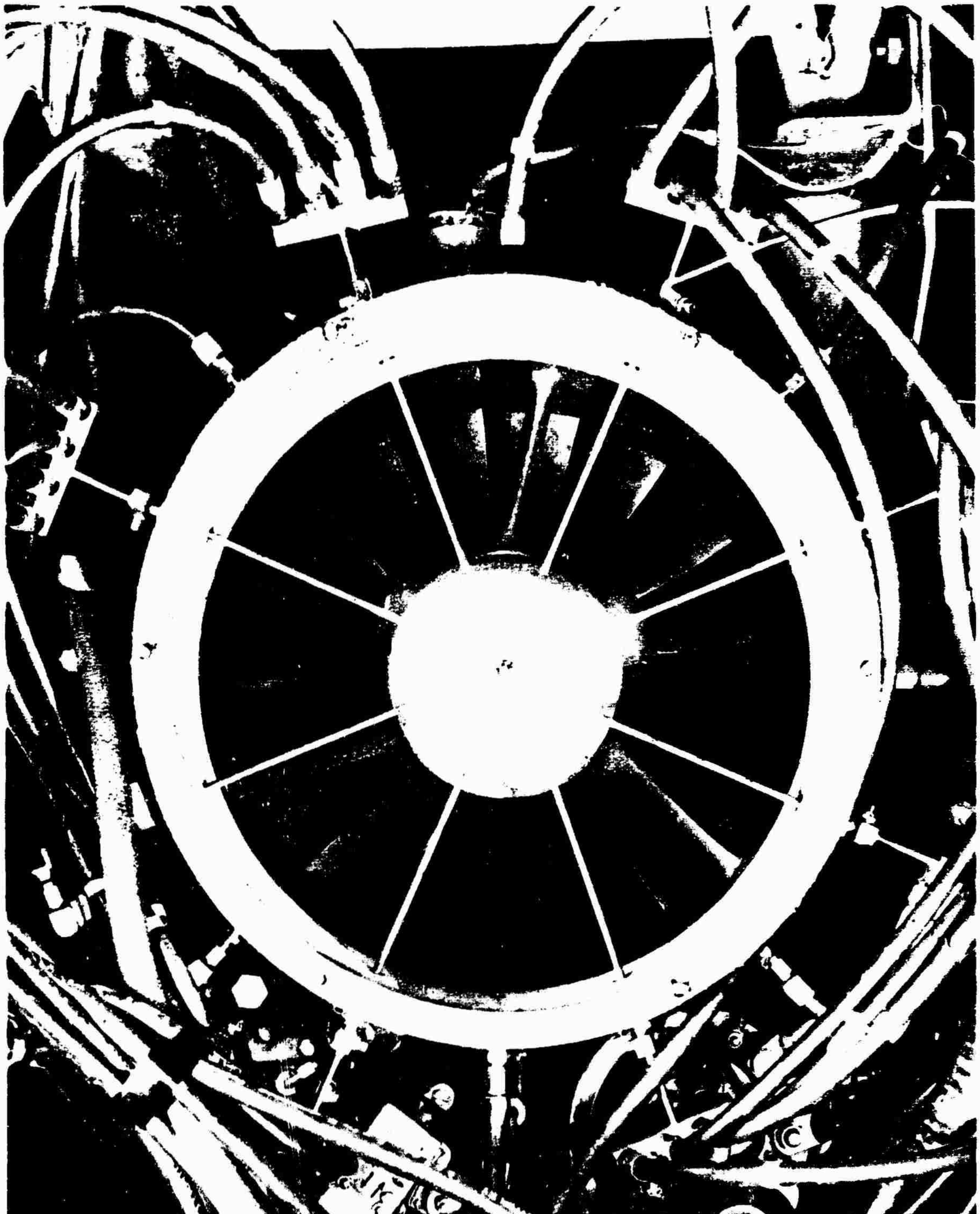


Figure 60. Installation of Adapter Ring - Model 357-1 Inlet Distortion Test.

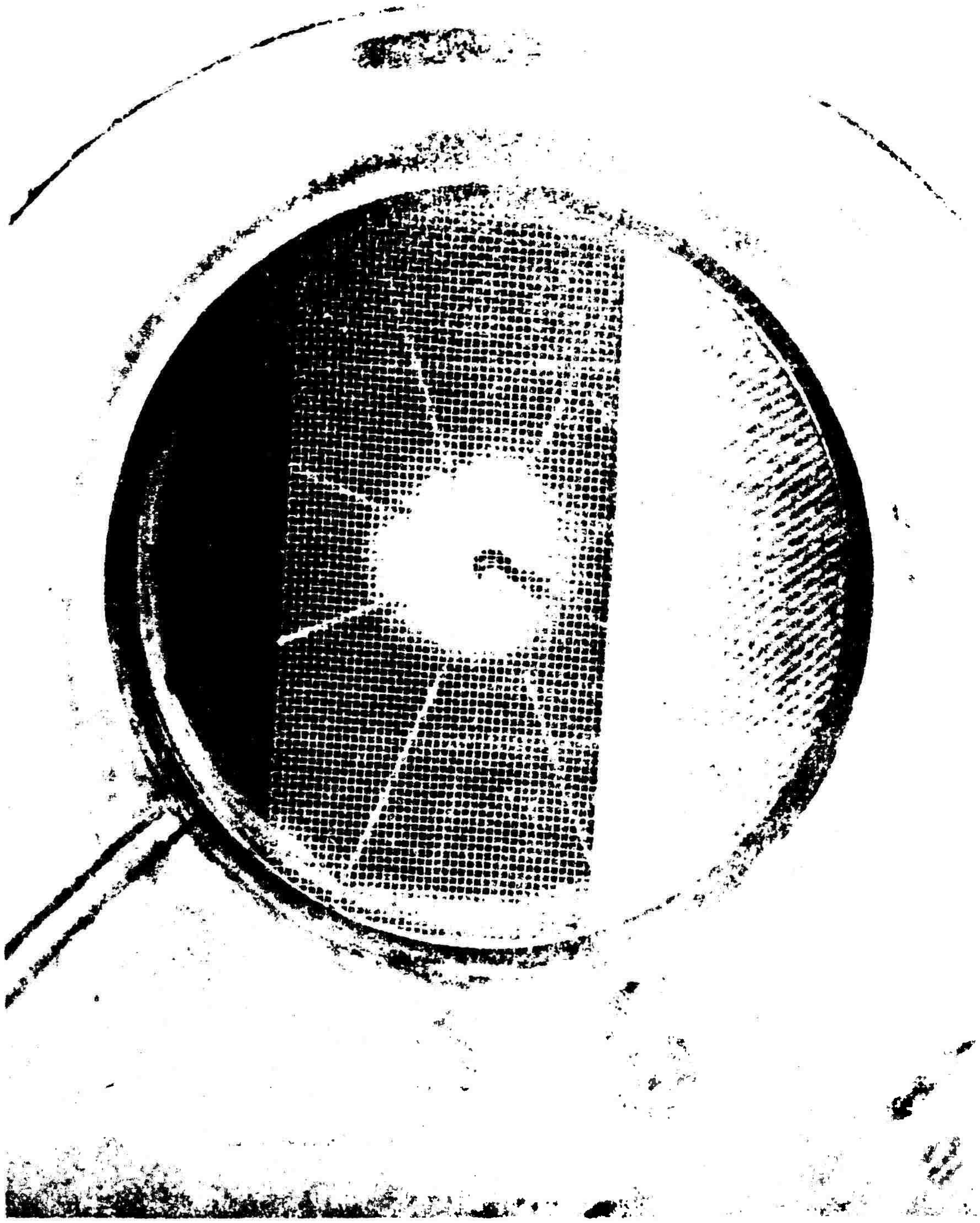


Figure 61. View of Distortion Screen in Model 357-1 Engine Inlet.

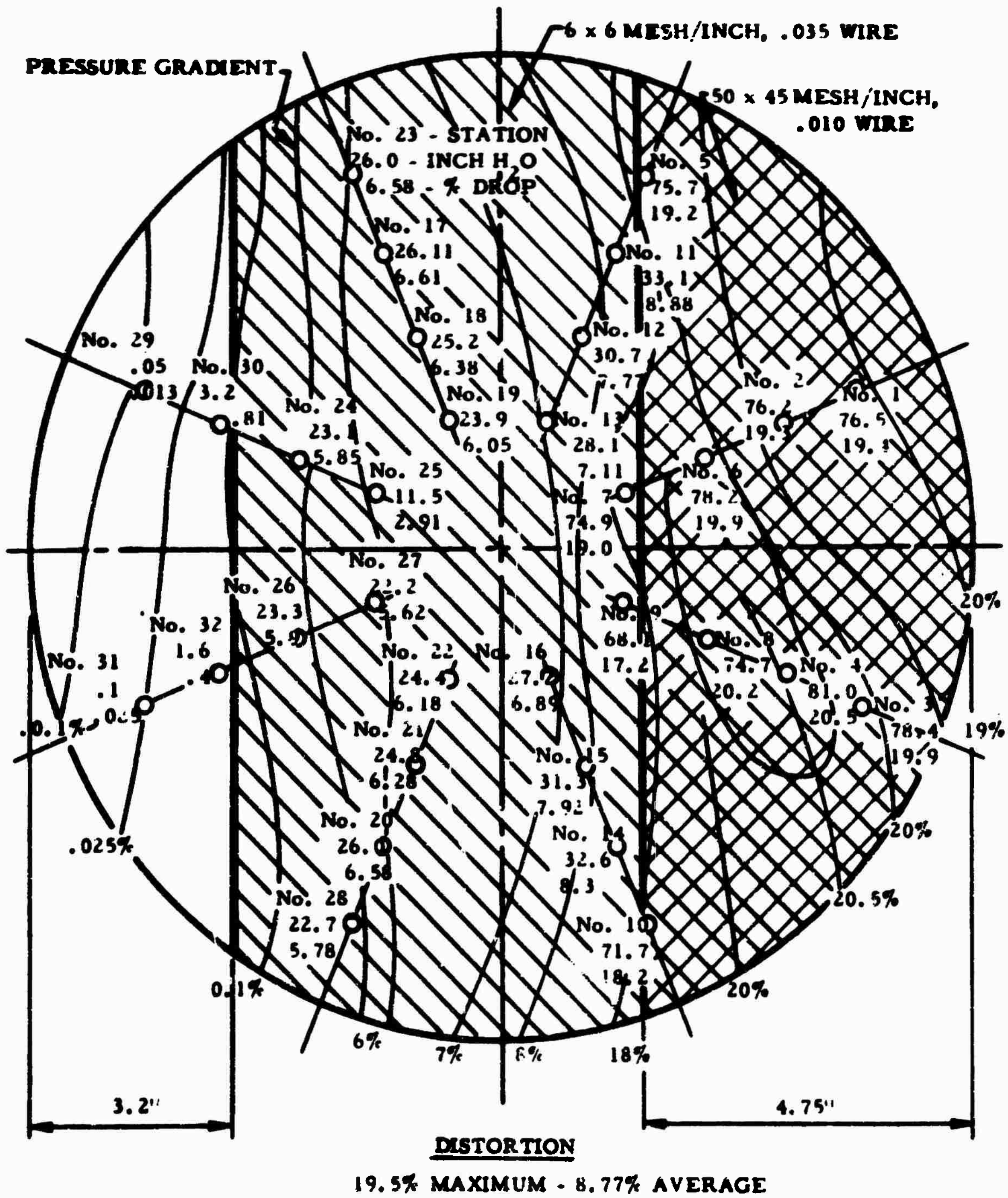
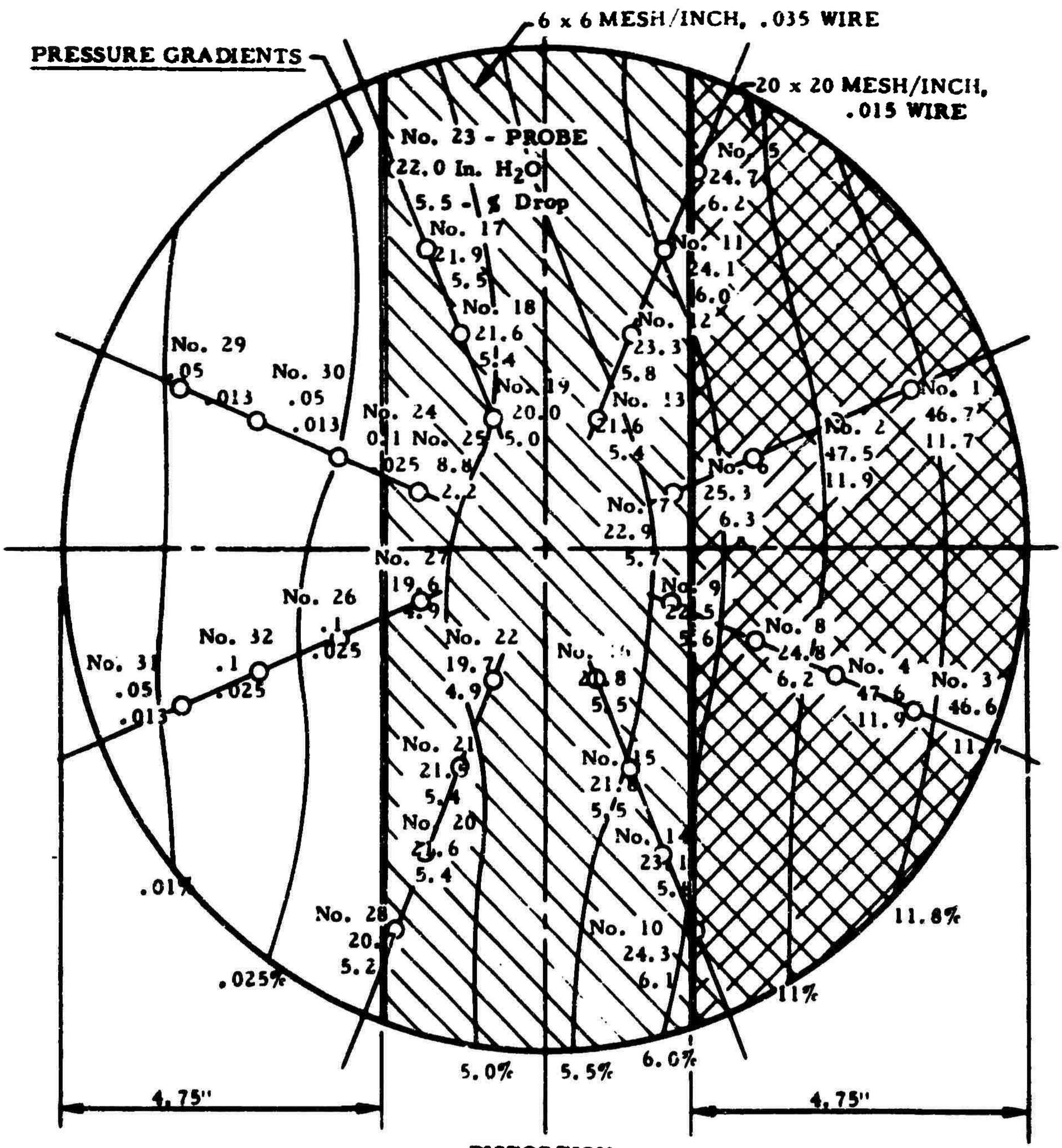


Figure 62. Model 357-1 Engine Inlet Distortion - Screen Pattern "A."



DISTORTION
11.8% MAXIMUM - 5.8% AVERAGE

Fig. 63. Model 357-1 Engine Inlet Distortion - Screen Pattern "B."

The change of performance from standard inlet and Pattern "A" distortion at 22,000 r.p.m. is as follows: 8.5-percent decrease in thrust; 4.5-percent increase in specific fuel consumption; 6-percent drop in compressor pressure ratio; 4-percent decrease in fuel flow; 6.5-percent decrease in airflow; and a 25°F increase in exhaust gas temperature.

The change in performance with Pattern "B" distortion at 22,000 r.p.m. produced a 17-percent decrease in thrust, 11.7-percent increase in specific fuel consumption, 11.5-percent decrease in pressure ratio, 7.9-percent decrease in fuel flow; 13-percent decrease in airflow, and a 50°F increase in exhaust gas temperature.

The loss in total pressure across the screened section was measured by total pressure probes behind the screen. This loss was then divided by barometric pressure to obtain the percent pressure drop for every probe location. Pressure drop gradient lines, in percent, were plotted for Figures 56 and 57. The percent drop for the open, partially screened, and densely screened section is the average of the probe readings for each respective area. The overall average pressure drop is the sum of all the readings divided by the total number of readings.

The change in the various parameters with change in inlet distortion are presented in Figures 64, 65, and 66. The thrust, airflow, fuel flow, and pressure ratio decreases as distortion increases while the exhaust gas temperature increases as the distortion increases.

After obtaining steady-state points with the distortion screens, the engine was accelerated from idle (11,000 r.p.m.) to maximum speed (22,000 r.p.m.) with no observable surge. A second acceleration with the higher distortion screens, Pattern "B," resulted in no surge during acceleration from idle to maximum r.p.m.

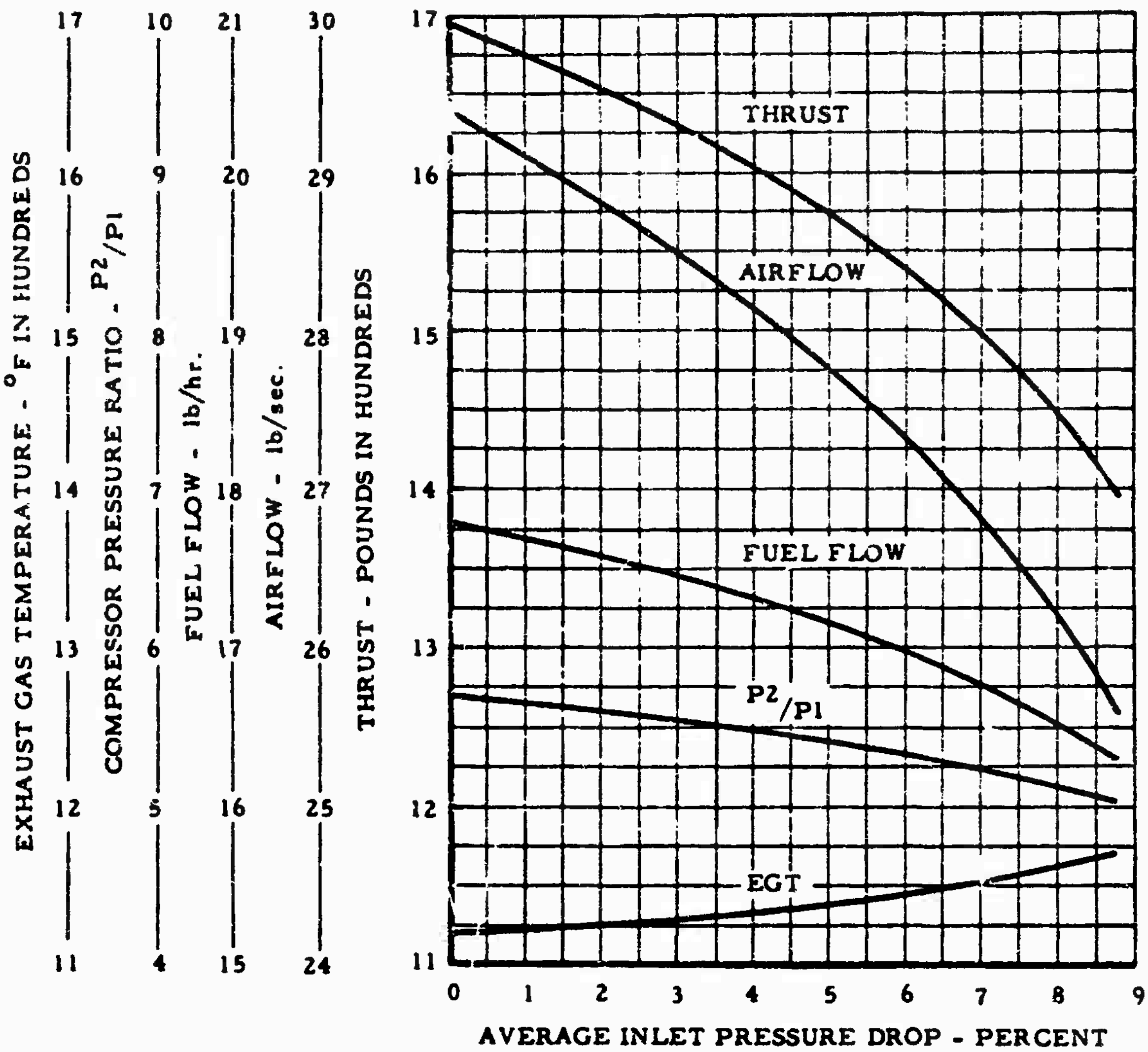


Figure 64. Parameter Change Versus Inlet Distortion at Design Speed - Model 357-1 Engine.

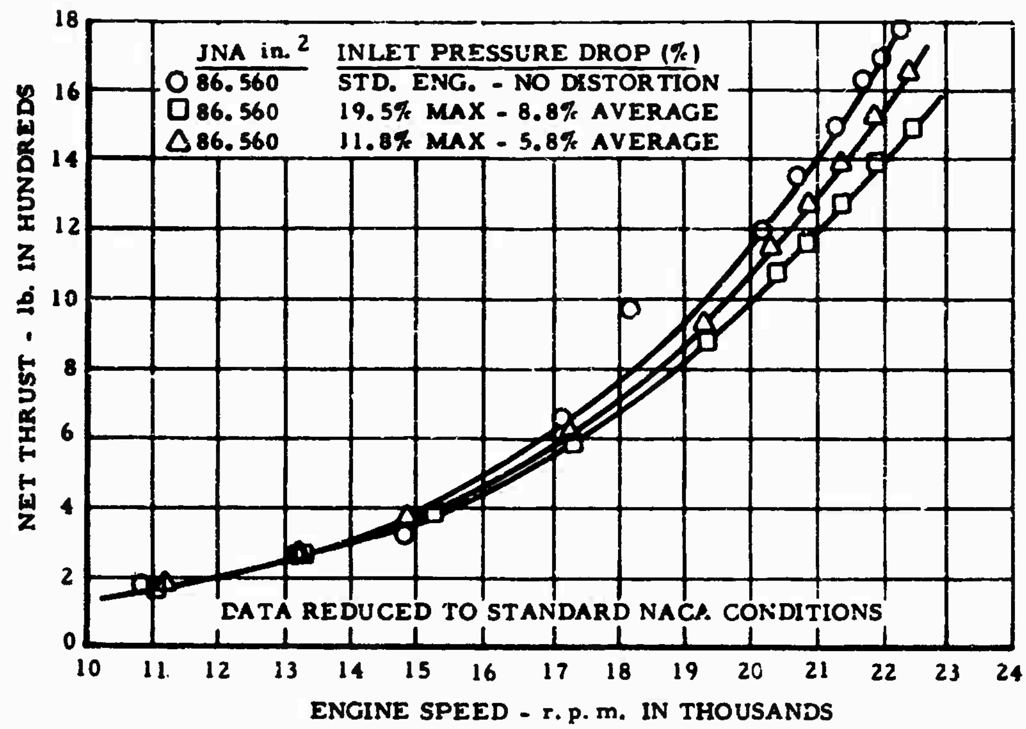


Figure 65. Effect of Inlet Distortion on Thrust - Model 357-1 Engine.

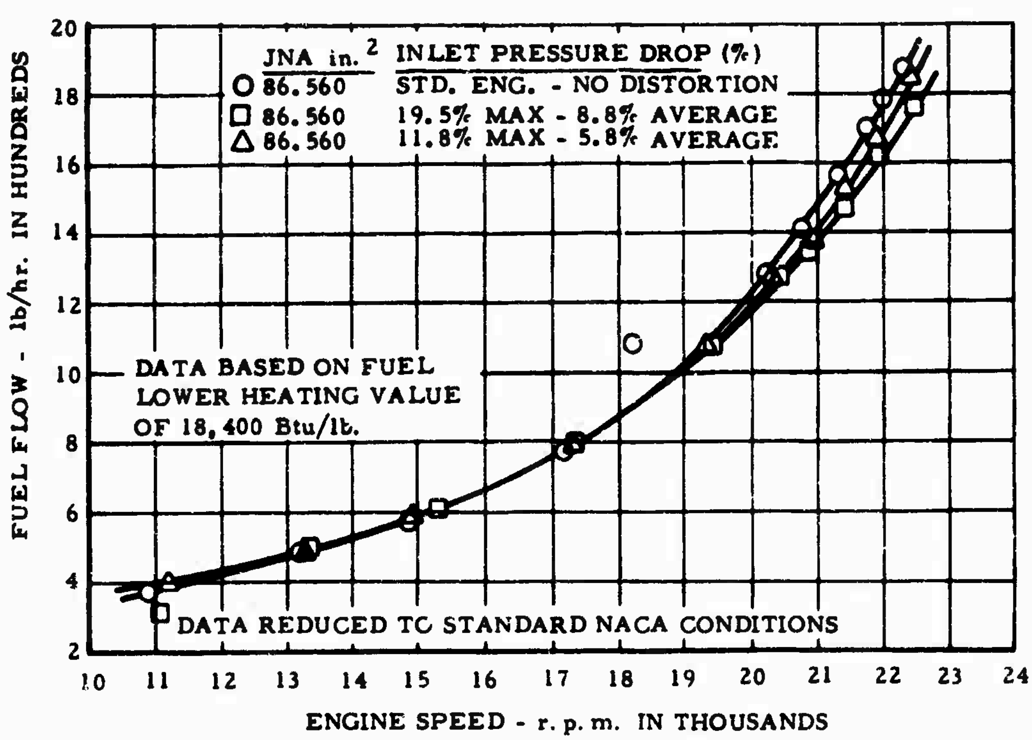


Figure 66. Effect of Inlet Distortion on Fuel Flow - Model 357-1 Engine.

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13. ABSTRACT

Volume XI of Heavy-Lift Tip Turbojet Rotor System discusses the design of the Model 357-1 engine for rotor tip operational environment. The detail design meets requirements of MIL-E-5007B except for items peculiar to rotor tip operation. These items include operational attitides, effect of continuous gyroscopic couples and high g field environment.

This volume includes an analytical design analysis of the static structure, rotating elements, bearings, accessory drives, lubrication system, and miscellaneous engine parts.

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